

**FINAL REPORT FOR THE UNMANNED MULTIPLE
EXPLORATORY PROBE SYSTEM (MEPS)
FOR MARS OBSERVATION**

Volume I. Trade Analysis and Design

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ABSTRACT

This report presents the unmanned Multiple Exploratory Probe System (MEPS), a space vehicle designed to observe the planet Mars in preparation for manned missions. The options considered for each major element are presented as a trade analysis, and the final vehicle design is defined.

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INTRODUCTION

In August 1987 a commission's report defining long-range goals for the U.S. space program was delivered to James Fletcher, chief of the National Aeronautics and Space Administration (NASA). (20:20-21) One proposed goal was the need for manned Mars explorations. To carry out this goal, unmanned vehicles would be used to observe Mars and gather data about the red planet; this data would then be incorporated into the design of a manned mission by 2010. (20:10)

This report details the description of an unmanned Multiple Exploratory Probe System (MEPS) to accomplish the necessary Mars observation leading to the manned Mars missions. Using surface rovers, orbiting satellite equipment, and a return vehicle, the knowledge of Mars will be greatly increased. Thus, design of manned missions would have a well-founded and well-defined starting point.

The MEPS mission will be completely described in this report. Construction in space and the materials of the vehicle will be discussed, followed by components of the transit flight to Mars--trajectory, guidance and control, and aerobraking. The modules comprising the vehicle, including the Command Information Center (CIC) and propulsion systems, will be described. Orbit and surface observations and data recovery (both telemetry and return vehicle) will round out the report.

MEPS MISSION DEFINITION

The unmanned Multiple Exploratory Probe System (MEPS) proposed for the detailed surface and atmospheric study of Mars has evolved into the following configuration. A full explanation of the MEPS vehicle and mission will be made in this report; a brief summary of MEPS is presented here.

The entire vehicle consists of five modules, separated by connection units (connectors), and an aerobrake. Some of the connectors and segments will be constructed together on Earth, and the remaining systems will be integrated in Low Earth Orbit (LEO) close to the Space Station. Each module contains a major vehicle component; these systems are classified the main propulsion, polar lander, secondary propulsion, CIC/satellite, and the equatorial lander (see Figure 1). Each module is a thin-walled cylinder 25 feet in diameter with the length defined by the respective module. The connectors, which are 25 feet in diameter and five or ten feet in length, will provide a quick, efficient mode of component integration in LEO following transport by Heavy Lift Launch Vehicles (HLLVs).

The vehicle will be staged to Mars. The main propulsion unit will provide the initial propulsive burn (ΔV) to place MEPS in a transfer orbit to Mars. Upon completion of this operation the unit will separate from the vehicle and, using a smaller ΔV burn, will fall into a high orbit about the earth while MEPS pro-

ceeds to Mars. At a predetermined position in the transfer orbit, the polar lander will separate from the vehicle and change course so that the lander can enter into a polar orbit about Mars. The remaining vehicle will move into a highly elliptical orbit about Mars via a propulsive maneuver from the secondary propulsion system. This initial orbit will be reduced and circularized by aerobraking through the Martian atmosphere, and the secondary propulsion system will adjust the final orbit following the aerobraking process.

Once orbit circularization is obtained, the aerobrake will separate from the vehicle to create an opening from which the equatorial lander will egress. After departure of the lander the satellite will be deployed into an appropriate orbit for Mars observations. Studies will also be conducted by the CIC and land systems; the collected data will be returned to Earth via the CIC. Most surface samples collected by the land rovers at both landing sites will be analyzed in the laboratories on board the landers. Other samples will be transported from each landing site into orbit about Mars to be returned to Earth orbit on a transfer vehicle. The satellite, CIC, and land systems will continue operations at Mars.

Figure 2 presents a pictorial representation of the mission profile.

STRUCTURES

This section documents detailed analyses of the MEPS vehicle performed from a structural standpoint. An initial finite element analysis is discussed and a description of the final configuration with information on new structural ideas is outlined. Proposed and suggested structural studies are also addressed.

Finite Element Analysis

A limited (25 node) finite element analysis was performed on the two initial truss configurations, designated back-slash and alternating-slash, using material information and various physical properties of the Space Station truss structure (reference 15). In the back-slash configuration, shown in Figure 3, the diagonals of a truss bay are arranged in the same direction; the cross in this figure appears because the diagonals on the backside of the truss are mirror images of the front diagonals. The diagonals of the alternating-slash configuration are organized in an alternating manner (see Figure 4). The initial analysis was conducted to determine exactly how the truss structure would deflect under some load condition. As seen in Figure 4, the alternating-slash configuration indicates, for a given loading, a contraction in the lower bay and an expansion in the upper bay; the center bay does not noticeably change size. For the back-slash configuration, the bays stay in the plane but the drift in the center of the bays hints at a rotation in the struc-

ture. Another view, Figure 5, shows more clearly the drift in the opposite direction for the top and bottom of the truss. Keeping in mind that the top nodes are located in the back, the opposite drift suggests a rotation of the structure. The rotation and out of plane deflection of the two configurations are not acceptable for a primary load carrying structure, especially when considering the mass this truss would add to the vehicle.

In an attempt to reduce some of the extra structural mass of the vehicle, a new configuration was proposed. This configuration used an outer skin constructed in space around the internal modules. The skin would be composed of many segments attached via existing connection techniques developed for the space station. Each outer skin segment was ten to twenty feet in length and, using six segments to completely enclose an internal module, would require 60 to 120 segments. In view of the amount of segments needed, in-orbit construction would be prohibitive when considering both the amount of extra-vehicular activity (EVA) time required by astronauts and the overall completion time. Thus a configuration (described in the next section) which details Earth-based construction was chosen because of the less EVA time required.

Modules

The modules to be used for the MEPS main vehicle will be thin-walled cylinders supported by bulkheads and stringers (Figures 6 and 7). The bulkheads will be located approximately every ten to fifteen feet. These bulkheads consist of rings with

'I' cross-sections around the interior circumference of the module (see Figure 8 for section properties). The stringers, which extend longitudinally, are arranged at ten degree increments circumferentially around the modules. For a conservative initial weight approximation, aluminum is chosen as the material for all structural components; this metal is denser than the materials (metal matrix composites and aluminum/graphite/epoxy) originally considered for these components. The structural mass of the vehicle is approximately 111,500 pounds. The length of each module, with corresponding structural and total mass, is shown in Table 1. (Note: the structural mass presented does not include the weight of propellant tanks.) Since some modules will experience the thermal cycling associated with aerobraking, a thermal control system will have to be developed; a system which will regulate the temperature without using conventional radiators is currently being assessed by industry.

An example of a possible method to secure payloads in the modules is shown using the equatorial lander rover (Figures 9 and 10). The actual development of a beam support system would be accomplished during the developmental stage of the project (between 1989 and 1993). The system will have several requirements: (a) proper support of the payload during pre-launch and launch activities; (b) sufficient structural strength during transfer and insertion into Martian orbit to prevent payload from dislodging and damaging crucial elements and/or the module; and (c) the ability to disengage connection on command from the CIC

and move the "beams" away from the exiting payload. The components internal to the modules will be attached to the stringers and bulkheads in the modules via this support system.

A detailed finite element analysis of the entire vehicle of connected modules could confirm the material chosen for the module shell, bulkhead, and stringers. Using elements for the cylindrical module similar to those shown in Figures 11 and 12, a finite element model of the entire vehicle can be constructed. The forces and accelerations obtained from the aerobraking and propulsion analyses would be used for a detailed dynamic analysis. The results of this analysis would also determine the required number and material composition of the pins to be used in the connectors.

The CIC/satellite module will have doors similar to the Space Shuttle bay doors to facilitate deployment of the satellite. These doors will be approximately 15 feet in length and will be stiffened to withstand the forces incurred during aerobraking. The opening of the doors and the deployment of the satellite will be controlled by the CIC. The doors will remain open to allow the CIC's power generation unit, a solar array panel, to be deployed.

The final module is the Equatorial Lander module. A ten foot connector, which must withstand the drag load during aerobraking, will be used to connect this module to the aerobrake.

Connectors

The modules of MEPS are fitted together with connector modules (connectors). The connectors provide a quick and simple method of constructing the vehicle in orbit. Special pins have been designed to provide for connection of the modules without tools and to allow a minimum EVA time. At the time of this writing, space construction is not commonplace; therefore, consideration was given to alleviate the complexity of construction.

The connectors must be strong enough to support the loads of rocket engine thrust during Earth departure and Mars capture, support the loads of aerobraking, and, with modules, resist bending and torsional loads. The connectors are modeled after the inner tank of the Space Shuttle External Tank. The axial loads will be resisted by the corrugated skin of the connector; due to the corrugation this skin design forms "built-in" stringers. The connector is not pressurized, but any circumferential stress will be resisted by barrel rings at the ends of the connector. The barrel rings also provide the attaching ring of the connector (Figure 13).

The attaching ring of the connector consists of the tang portion of a tang and clevis joint (see Figure 14). Several holes are located around the circumference of the tang and clevis sections to allow the attach pin to be locked to the clevis joint. The end of the tang is rounded and the clevis is beveled

to allow easy alignment of the connection. The seats of the components are slotted in order for the holes in the clevis and tang to line up properly.

The attaching pins have been designed to provide ease of operation and sufficient strength to maintain the structural integrity of the connector. The pin consists of a "T-handle" grip with a threaded inner bolt that rotates inside the outer sleeve. The outer sleeve of the pin has a lightweight attach clip which connects to the outer surface of the clevis ring; this attach clip allows the inner bolt to be rotated independently of the outer sleeve. As the "T-handle" is rotated, the inner bolt revolves to expand the leaves of the pin on the inner surface of the clevis ring. The pin securely fastens the module to the connector while locking into position; the inner expansion sleeves, the attach clip, and the T-handle insure fixed pin placement.

Each module will be launched into orbit with a connector attached to the forward end. Thus, when each module reaches the construction site only one module needs to be connected. The construction is to be performed at the Space Station, which is assumed to be operational. The remote manipulator system of the station will be used to help place the modules into position.

COMMAND INFORMATION CENTER

The Command Information Center (CIC) is the "heart" of MEPS; all system functions and data collection/processing will be performed by the CIC. The center will be composed of computers, communications and observation equipment, and a solar array panel. The CIC has a great deal of responsibility, and the success of the MEPS mission depends on the ability of the CIC to manage the variety of tasks put upon the unit.

Computers and Flight Software

The computers in the CIC will have to be the most up-to-date systems that are available when construction of MEPS begins. The computers must be able to process all of the flight software specially written for MEPS. Guidance and control, to be discussed in a separate section, will also be handled by the computers. In addition, some degree of artificial intelligence is desired to take care of any problems that may arise. This capability would help alleviate the problem of the time lag for communications between Mars and Earth.

Flight software will be required for many different tasks. At the very start of, and throughout, the mission the computers will run checks on all systems to insure successful operation. The trajectory for Earth-Mars transit must be maintained during the transfer; propulsive burns necessary for the transfer and

orbit circularization will be controlled from the CIC. In addition, orientation of the communications antenna and solar array panel will be monitored using software packages.

Communications

Two communications links will be required by the CIC; these links operate in the narrow and wide bands. The links will enable the CIC to perform a great number of tasks, including Mars observations, Earth-MEPS communications, and data processing. Prior to leaving Earth orbit a parabolic antenna will be deployed for continuous contact during transit.

The first link operates in the narrow band (S-band). The S-band link provides telemetry and is used for general house-keeping. On-board flight control and critical function monitoring is carried out over the S-band link, as well as storage and issuance of commands and data retrieval/processing. The second link is in the wide band (Ku-band); the wide band is used for the collection of all Mars observation data. (2:74,76)

CIC communications will be connected with three elements. One link will be made between the CIC and the satellite to transfer observation data collected in the higher satellite orbit to the data collection system of the CIC. The surface rovers will communicate with the CIC to relay data and accept overriding commands from Earth scientists. Finally, MEPS must remain in contact with Earth so that all observation data can be relayed to Earth ground stations, and commands and instructions can be sent to MEPS.

A parabolic antenna will be deployed from the CIC module. The antenna will be constructed of a graphite-epoxy sandwich construction, with graphite-epoxy struts for feed supports; the size of the antenna will be left for later analysis using appropriate computer techniques. At pre-set times, the antenna will be directed toward the Martian surface or the observation orbit to make contact with the separate systems. Once the data collection files are completely full, or at pre-programmed times, the antenna will be pointed toward Earth and the data will be relayed to receiving stations.

CIC Observations

In addition to the satellite, the CIC will conduct observations of Mars; the collected information will enhance the present knowledge of the Mars surface and atmosphere. Three experiment packages similar to instruments used on previous Earth satellites will be utilized.

The observation and mapping of the Martian surface will be conducted by a system similar to the Landsat-D Multispectral Scanner (MSS). In its present use, the MSS captures Earth images in the spectral bands ranging from the visible to the near infrared. (10:350) Applying this system to Mars will allow Mars mapping and surface scanning.

To measure the heat budget of Mars, the Electrically Scanning Microwave Radiometer (ESMR) of the Nimbus project will be used. This experiment will map the thermal radiation from the surface of Mars and the atmosphere by using microwaves to

sense the temperature in the scanning area. From the scans, a thermal image of the entire planet of Mars will be available.

(10:80,89)

An experiment for the study of the environment surrounding the CIC is based on the Tiros Satellite Environment Monitor (SEM). The SEM is a three-instrument multidetector unit that monitors solar particulate energies; these energies fall in the solar proton, electron flux-density, and alpha particle spectrums. (10:66) Investigation of space during transit and while in orbit about Mars is important to manned mission design so that any anomalies deemed dangerous to man may be discovered and analyzed.

Solar Array Panel

A solar array system will provide the power input for the CIC, both directly and indirectly. The experiment packages, computers, and other systems will obtain power from the panel while MEPS is in contact with sunlight; when MEPS crosses into the dark side of Mars batteries will supply the necessary power.

For convenience in deployment of the array system, a solar panel is selected. The panel will be contained in a box located in the CIC module between the CIC systems and the observation satellite. Similar to the solar array mission of the Space Shuttle Discovery (Mission 41-D), the panel will be unfolded to extend into space. The panel will be oriented toward the sun.

The panel will be composed of conventional silicon solar cells, which have been continuously used for space applications.

The cells have an efficiency (ratio of power output from array to power input from sun) of 9.6 percent, and are not very thick (0.0079 to 0.0157 inches). (21:165-166) At the start of operation, the available power from the array is designed to be approximately 1500 watts; this requirement should allow sufficient power to the MEPS systems and recharging of the batteries. Due to exposure to ionizing radiation and changes in orbit inclination the power will decrease to 1100 watts by the end of the five-year mission. (2:88) From these power considerations, the total area of the array is determined to be 110 square feet. (21:123)

A thin coating over the cells is desired for three reasons: protection from space particles and radiation, enhancement of the amount of sunlight to reach the solar cells, and improvement of the quantity of heat rejected from the cells. For the conventional silicon cells the recommended coating is silicon monoxide. (21:166,242)

Deployment will be accomplished by a special mechanism designed for this array. Another specially designed control will monitor the solar array direction throughout the mission and maintain the array perpendicular to the sun's rays; correcting the position of the array will allow maximum solar collection.

When the vehicle is out of direct sunlight on-board batteries will be used to supply power to the systems. To maintain the performance of the systems two 50-ampere hour (AH) batteries will service the satellite. Although the 50-AH battery has a

greater mass than other possibilities (such as a 20-AH battery), the 50-AH battery has a much better performance. (2:90) For the MEPS mission, the best performance possible is required of all components since satellite repair will be impossible.

Mission Preparation

Preparation for different phases of the MEPS mission will be regulated by the CIC. The central computers will be programmed with the proper information to begin all propulsive burns for the transfer to Mars and the orbit capture about Mars. The aerobraking phase will require system checks by the CIC before and during aerobraking. Finally, deployment of the satellite and surface rovers will be accomplished with programs in the flight software.

TRAJECTORY FOR EARTH-MARS TRANSIT

The most important component of the MEPS mission analysis is the trajectory for the flight between Earth and Mars. Based on the propulsive burns required for velocity changes, the mass of the propellant and thus the initial mass leaving Earth can be determined. In addition, the time required for the transit can be calculated.

The determination of the precise path of interplanetary travel requires numerical integration of the complete equations of motion. For preliminary mission analysis, however, the "patched-conic" method of approximation can be used. The patched-conic method consists of elliptical paths between planets and hyperbolic passages near the target planet; these elliptic and hyperbolic paths, or conic paths, are "patched" together to construct the interplanetary trajectory. In addition, the method allows the use of two-body motion; a planet's gravitation is turned on and off in relation to the distance between the vehicle and planet, and no other body will have an effect on the vehicle.

Ecliptic Plane Consideration

Because the MEPS vehicle will be constructed near the space station, the inclination of the station's orbit must be considered. The ecliptic plane is the reference plane in which all of the planets rotate about the sun. Relative to the ecliptic, the space station is in an Earth orbit inclined twenty-eight and

one-half degrees. The proper trajectory for transfer from this orbit to Mars would require an inclination change before Earth orbit departure; this inclination change would increase the initial mass by increasing the fuel requirements.

To avoid the inclination change, orbital transfer vehicles (OTVs) will be used to change the orbit inclination while MEPS is still in Earth orbit; MEPS will be moved to the ecliptic plane so that the transfer can be completed in a single plane. Thus, no additional fuel will be required by MEPS. Upon reaching Mars, the vehicle will remain in the ecliptic plane for ease of deployment of the equatorial lander and complete satellite observation.

Trajectory Description

The MEPS vehicle will begin its journey from a 270 nautical mile orbit above the Earth's surface (see Figure 2). Once all system checks have been completed, a propulsive ΔV burn will be performed. The position in the circular orbit where the burn will be made is at the farthest point from the sun; this burn will place MEPS into an elliptic transfer orbit about the sun. The Earth's gravity will continue to have an effect on the satellite until the Earth's sphere of influence is exceeded, at which point the effect of the sun becomes dominant. The vehicle will continue in the elliptic orbit until the approach toward Mars.

Once the sphere of influence of Mars is reached, the sun is "turned off" and Mars is "turned on". At this instant the vehicle is on a hyperbolic path around Mars. Using a propulsive

burn at the closest approach of the near-Mars passage, an elliptic orbit about Mars can be achieved. This orbit is the system check-out orbit prior to aerobraking.

The proposed transit will require minimum energy and is known as a Hohmann transfer. Minimum energy equates to maximum time for the transfer; the required time to begin the transfer from a circular Earth orbit and arrive in the elliptic orbit about Mars is approximately 285 days.

Table 2 shows the relevant information concerning the Earth-Mars transit of MEPS.

PROPULSION SYSTEMS

The propulsion system will be a major element of the MEPS vehicle. Propulsive burns will start the transfer to Mars, allow capture into an elliptic orbit about Mars, and assist in the circularization of the MEPS observation orbit.

Background

For the initial analysis several systems emerged as possible candidates for MEPS propulsion. Considered for application were nuclear, electric, and chemical propulsion. The choice for a chemical propulsion system resulted from a careful study of the alternatives.

From a technological standpoint the nuclear system is very attractive; much less mass would be required and the system has a higher specific impulse. (12:366) However, nuclear propulsion is probably the most controversial and the public and government leaders might resist such a system in space. Additionally, nuclear propulsion technology is not advanced enough for interplanetary travel. (16:51)

An electrical system is also attractive but, like nuclear propulsion, the potential for large space vehicle applications has not yet been reached. The electrical system has a higher specific impulse than either chemical or nuclear systems. (12:280) However, the power supplies which must accompany an

electrical system are large, heavy, and typically inefficient, and the low thrust levels have limited current applications to spacecraft control. (25:9-11)

The liquid chemical propulsion system was chosen for the system's maturity over any other type of rocket engine. The cryogenic propellant combination of liquid hydrogen and liquid oxygen will be used for Earth departure. The storable propellant combination of nitrogen tetroxide and monomethyl hydrazine will be employed for all other propulsive maneuvers.

Earth Departure

The greatest amount of thrust during this mission is needed for LEO departure. Liquid hydrogen (LH_2) and liquid oxygen (LO_2) were chosen primarily for the high specific impulse that is obtainable with this fuel. Experience with this combination in previous space applications also contributed to the selection.

A key comparison point for LH_2/LO_2 engines is the nominal thrust level. From an analysis of burn times desirable high thrust levels were verified. Four LH_2/LO_2 engines were compared- the RL-10, the J-2, the Space Shuttle Main Engine (SSME), and the Space Transportation Main Engine (STME). The RL-10 was removed from consideration because twenty-nine engines would be necessary to obtain the required thrust. The J-2 has much experience but the system is based on 1960s technology. The SSME produces the highest thrust level of all four engines but has a high

cost and low level of reliability. The logical choice is the STME; the engine is being designed for reusable space applications and optimum space operation.

The STME is a gas generator cycle engine, a primary distinction from the staged combustion cycle of the SSME. Although considered to be a "step backwards in technology" from the SSME with a lower thrust and lower specific impulse, the engine is projected to be more reliable and cost-efficient. The STME is still in the development stage, so present engine characteristics are only projected values. From the performance analysis of the engine the burn time and propellant volume and mass may be obtained. These values as well as some common operating parameters of the STME may be found in Table 3.

The total vehicle mass of 381,500 lbm will require a propellant mass of 472,520 lbm to travel to Mars, assuming no boil-off prior to the Earth-Mars transit burn. However, for the vehicle assembly period and the Earth to LEO transit, boil-off of the cryogenics must be considered. The boil-off is a function of time and was taken to be 12.0 lb/hr for LO_2 and 18.0 lb/hr for LH_2 (30:99); the required mass of propellant will increase to 494,120 lbm.

An optimization study on the required number of stages revealed that only one stage of a single STME would be needed for LEO departure. This stage will propel the entire vehicle to the

Earth-Mars transfer speed and then separate from the main vehicle. The spent stage will be deboosted by small, solid retro-rockets and enter into a highly elliptic orbit about the Earth.

Secondary Propulsion System

A second propulsion system will be required after the main stage has been released. The second stage will allow MEPS to be captured in Mars orbit in preparation for aerobraking and will also be used to help circularize the final orbit about Mars. The secondary propulsion module is located ahead of the polar lander-rover module; the lander will separate before operation of the engines.

Storable propellants will be used for the secondary stage. The use of storables will be necessary to avoid the "boil-off" problems that occur with the use of cryogenic propellants on missions of extended length. (4:95) Monomethyl hydrazine (MMH) and nitrogen tetroxide (N_2O_4) will be used as the fuel and the oxidizer, respectively, because of high performance capabilities. (13)

A specific engine cannot be specified at this time because no existing engine will use storable propellants and provide the required thrust. However, the propellant and mass requirements are specified so that an engine may be developed for these purposes. For this analysis, the Space Shuttle Orbiting Maneuvering System (OMS) engines are used for performance standards.

The second stage must provide four burns for this mission. The first burn will enable MEPS to be captured in Mars orbit to

begin aerobraking. The second, third, and fourth burns will be necessary in order to lower and raise the periapsis and adjust the apoapsis, respectively, for orbit circularization before and after the aerobraking process. Table 4 presents the magnitudes of the propulsive burns and the propellant mass requirement for each burn.

Using the values from the table the total propellant mass required for the second stage is 65,026.35 lbm. For this mass two spherical tanks are needed to contain the propellant (MMH in one tank and N_2O_4 in the second). The tanks will be ten feet in diameter, which is slightly larger than required. The extra fuel will be retained as a safety factor in case of emergency propulsive maneuvers.

The two spheres will be contained in a ten foot long module. Three engines will also be located in this module; two engines will pivot out from inside the module to be directed towards the front of the ship at a small angle from the longitudinal axis, and the third will be positioned in the rear and directed rearward (see Figure 15). The engines with forward exhaust are contained within the propulsion module in order to maintain the continuity of the twenty-five foot diameter cylindrical module arrangement.

The secondary propulsion module is located at the aft end of the ship and the aerobrake is at the front; however, the retro-thrust must be provided by engines pointing forward. The two forward-directed engines mentioned above will tilt outward from

within the propulsion module for this purpose. The third engine (aft-directed) will provide the forward thrust used to circularize the orbit.

For the analysis of propellant mass for these engines, the tilt angle was assumed to have a negligible effect on the deceleration capability. However, when the thrust level of the engine is being determined the tilt angle must be taken into account because the actual thrust available for deceleration will be reduced by a factor of the cosine of the tilt angle.

GUIDANCE AND CONTROL

All guidance and control systems are composed of four system elements: sensing components, references, control logic and electronics, and actuators. The guidance and control system components defined in this section are based upon related material in the AFSC Design Handbook for Space Vehicles.

Sensing components use physical mediums to give a system information concerning position, state, and condition of that system with respect to the available medium. Both inertial sensors and optical sensors were considered for this mission. Inertial sensors use Newton's laws of motion to sense motion with respect to an inertial reference. MEPS will employ two types of this sensor--gyroscopes and accelerometers. Gyroscopes sense angular rate or displacement with respect to an inertial reference by the use of the properties of a spinning rotor. Accelerometers utilize the inertial reactions of a proof mass to determine angular and linear acceleration.

Optical sensors utilize optical searching of space to detect and track celestial objects or obtain a reference from a body's infrared horizon. Two types of optical sensors will be used--a star sensor and a horizon sensor. Star sensors search a portion of space for the purpose of detecting and tracking a celestial body; horizon sensors use the infrared horizon of a planet's surface to obtain a reference. Three horizon scanning techniques

were considered: conical scanning, edge tracking, and radiometric scanning. Conical scanning uses a small field of view rotated through a large cone angle; edge tracking uses a small field of view locked on to the infrared horizon; and radiometric scanning uses three different lines of sight which are fixed on three different points on the horizon to observe infrared properties of the horizon.

MEPS will be using inertial reference systems throughout the entire mission. Inertial reference systems are reference systems which maintain a fixed coordinate system in inertial space. Two types of inertial references will be utilized for the Mars mission. The first type is the gimballed platform reference (Figure 16). This reference system contains a stable element which is inertially fixed and gyro-stabilized and is totally isolated from the rotational motion of the ship's structure by the use of gimbal rings. Any angular motion is sensed by the gyros and a counter-torque is applied by the control system of the vehicle to restore the craft to the original state. The second type of inertial system is the body bound reference which, like the gimballed platform system, uses gyros to sense angular motion. However, for this reference a computer digitally interprets the gyro output signals. The body-bound system, therefore, maintains the reference axis electronically while the gimballed platform performs the same function mechanically.

The purpose of control logic and electronics in a guidance and control system is to evaluate the signals from the sensors

and references in order to operate the actuators which control the vehicle. Actuators supply the torques necessary to change or maintain the attitude of the spacecraft as a result of external and internal disturbances. The two categories of actuators are expelled-momentum actuators and momentum-storage actuators. Expelled-momentum actuators use a "controlled dumping of mass" to provide the force by the control system; momentum-storage actuators use inertially fixed reaction wheels (spinning disks) and the principal of conservation of angular momentum to control the vehicle.

In choosing a guidance and control system for MEPS four primary phases of the mission must be considered: Earth-Mars transfer, aerobraking, Mars orbit, and return transfer. Each of these phases will require specific types of sensors.

During the Earth-Mars transfer a combination of inertial and optical sensors will be used. A star sensor will be employed as the main sensor for reference determination due to a greater accuracy over horizon sensors. However, a horizon sensor will be employed as a secondary sensor using radiometric scanning techniques; this technique produces greater reliability due to a lack of moving parts. Gyroscopes and accelerometers will also be used as secondary or backup sensors for this phase.

The second phase of the mission is the aerobraking phase. During the braking process optical sensors cannot be used due to ionization. Therefore, reference determination will be performed exclusively by the gyroscopes and accelerometers.

For the Mars orbit the star sensor and the horizon sensor will again be used to obtain a reference and maintain the elliptical orbit. Gyroscopes and accelerometers will be utilized as secondary sensors.

The final phase for which guidance and control is required is the return transfer from Mars to Earth. The star sensor, horizon sensor, gyroscope, and accelerometer will once again be employed for reference determination. A transponder must also be utilized in the return vehicle. This transponder will enable the return vehicle to track, rendezvous with, and retrieve canisters which have been boosted from the surface landers.

One primary reason for using both optical and inertial sensors is that gyroscopes and accelerometers have an unacceptable degree of error. An optical sensor can be utilized to monitor the inertial sensors for compensation of inertial sensor error. Thus inertial and optical sensors can be considered as a means for short term and long term stability, respectively, where stability is a measure of a component's resistance to error in measurements.

The remaining three components of the guidance and control system--the reference system, the control logic and electronics, and the actuators--will be consistent throughout all phases of the mission. That is, the types of components will not vary between phases.

The MEPS sensors will utilize both the body bound reference and platform reference systems. A body bound reference exhibits

a longer continuous operation time with fewer errors than the platform reference system, and will be employed as the main reference system. The platform reference system will be used as a secondary system because of a greater accuracy for short term usage.

The type of control logic and electronics which will be used for MEPS will be determined by considering the software available at the time of the mission.

The actuators which will be used for MEPS will be a combination of both expelled-momentum and momentum-storage actuators. Momentum-storage actuators absorb only cyclic disturbances (e.g. gravity gradients and aerodynamic forces). Expelled-momentum actuators are required for attitude adjustment due to the non-cyclic disturbances (e.g. thrust transients, fuel sloshing in tanks, etc.)

AEROBRAKING SYSTEM

A key feature of the MEPS mission is the testing of different systems proposed for a manned Mars mission. One of these features is the use of aerobraking to achieve orbit about Mars. Aerobraking is the process of obtaining a circular orbit about a planet by gradually reducing the vehicle's elliptical orbit through atmospheric passage. During each passage the vehicle experiences drag; this opposing force causes a reduction in orbital velocity which results in a decrease of the apoapsis distance while only slightly affecting the periapsis distance. Continuation of the process of entering and leaving the atmosphere eventually allows a circular orbit about the planet to be established. (9:397-398)

Aerobraking is a relatively new technique and has several advantages. First, the process eliminates the need for the large quantities of fuel which are required when retrorockets are fired. By utilizing successive passes through the atmosphere, the operation, and thus the technology, remains relatively simple. In addition, large structures, even if not aerodynamic and of great mass, are able to enter into a low circular orbit. (8:245-246)

The benefits of aerobraking are offset by several disadvantages. The amount of time required to obtain a circular orbit can be as much as two to three months. A second problem is great

variations in the rate of orbital decay. (9:398) Finally, at present the aerobraking process is only theoretical; no flight tests are available, but models have been developed on computers.

The following sections will detail the proposed aerobrake for the MEPS mission. A description of the aerobrake will be given, including the materials and construction. Then, the aerobraking process will be discussed.

Description

The proposed aerobrake is shown in Figure 18. The aerobrake is a 70 degree right circular cone with ribs and struts for support during braking. Rigid and flexible sections compose the surface of the brake. The 70 degree cone angle is not arbitrary; this angle was used for an aerobrake on Viking in 1976, and theoretical studies by NASA and private industry start with a seventy degree angle. (29) A ring section behind the brake provides attachment points for the struts in addition to providing a transition to the main MEPS vehicle.

The size of the aerobrake can be obtained by using Newtonian methods. Newtonian theory states that at high speeds (such as hypersonic) the forces on an object in the flow can be examined on the molecular level; the impact of each flow molecule produces the aerodynamic forces on the object. Using this approach, a drag coefficient for the brake can be determined. Next, using the ballistic coefficient $[W/(C_d \cdot A)]$ of the aerobrake for Mars

capture (28:16) and a given weight, the diameter of the aerobrake is calculated. All of the parameters of the proposed aerobrake are presented in Table 6.

The ribs will provide structural support and flutter resistance to the flexible region of the brake during aerobraking. The support struts will help to maintain the shape of the brake by keeping the brake in its original position. To minimize the sagging of the flexible portion between the ribs during braking, a solid structure will be positioned at the outer radius of the brake.

Materials and Construction

Developed through research by private industry, the brake will be composed of two different materials. The inner 25 foot diameter will be constructed of a rigid surface insulation (RSI) made of graphite polyamide honeycomb and ceramic foam tiles. The remainder of the brake will be a flexible surface insulation (FSI). The FSI will be a blanket of sheets of interwoven ceramic fibers (Nextel 440 and Nicalon) alternated with ceramic felt filler (see Figure 18). The weights for the RSI and FSI are given in Table 5.

The Nicalon silicon carbide fiber cloth is used on the forward surface of the brake. A thinner gauge of this fiber is used for an "interior woven truss network structure" to help retain the felt material. Nextel cloth is used on the back side of the brake with an RTV silicone coating; this combination will help to seal any openings and prevent "hot gas flow through the

composite fabric structure." The ribs and structural supports for the brake are constructed of graphite polyamide. (27:186)

An HLLV can be used to carry the center rigid section of the aerobrake to LEO. Because the remainder of the brake is flexible, the FSI and rib/structure support combination can be folded for the transit to LEO. If proper connections of the FSI to the RSI cannot be made in LEO, the entire aerobrake must be carried on an HLLV. For this case, folding the FSI over the RSI will allow easy transit.

Aerobrake Process

Aerobraking can be used in place of a propulsive burn to obtain an orbit about the target planet. Because aerobraking is not proven technology, however, a propulsive burn will be used for the MEPS mission, and aerobraking will be performed to circularize the orbit.

Once MEPS is in an elliptic orbit about Mars the on-board computers will make checks to insure that the vehicle is secure in preparation for aerobraking. These checks will continue for 1.5 orbits. When the vehicle is at the apoapsis of the second orbit a V burn will be applied to lower the periapsis of the orbit into the Martian atmosphere. As MEPS continues in orbit the Martian atmosphere will eventually be reached. At this time atmospheric drag will begin to act on the aerobrake, slowing MEPS. The change in velocity is used to calculate a shorter semi-major axis length for the orbit. This new semi-major axis yields a new apoapsis length, assuming that the periapsis will

remain at the same altitude. Several passes through the atmosphere will reduce the apoapsis sufficiently to circularize the orbit about Mars.

Several assumptions must be made in the consideration of aerobraking. In the analysis of the braking the drag on the vehicle is assumed constant and is determined using the atmospheric density and orbital velocity at the periapsis. This determination of the drag is not completely accurate, since both the density and velocity will be changing during the passage through the atmosphere. However, the assumption to calculate the drag at periapsis does give a good first approximation for aerobraking parameters. The assumption of constant drag is being considered by private industry. Accelerometers on the brake will measure the decelerations of the vehicle and compare with calculated values from flight software. Any deviations from the desired deceleration will result in small, accurately-measured V burns to either slow down or speed up the vehicle to obtain the desired deceleration.

Because the vehicle is traveling through an atmosphere the stability of the vehicle must be analyzed. The main reason for using a 70 degree cone aerobrake now becomes clear--this cone is very stable. The location of the center of pressure is approximated to be one brake diameter aft of the vehicle's nose. Provided the center of gravity is located forward of the center of pressure the stability of the vehicle is ensured. (27:183) In

addition to the brake's inherent stability, momentum wheels will be incorporated to provide a zero angle of attack during the passage. (8:247)

When the apoapsis altitude is reduced to the desired value the aerobraking process is complete. When the vehicle reaches the apoapsis of the orbit a ΔV burn will be applied to raise the periapsis back to 270 nautical miles (the proposed MEPS orbit). If the actual apoapsis altitude is below the desired altitude, a final ΔV burn will be made at the periapsis to raise the apoapsis. Upon completion of this burn the vehicle will be in a circular orbit about Mars at an altitude of 270 nautical miles.

An analysis was conducted to obtain various parameters of the aerobraking process. For a vehicle weight of 200,000 lb aerobraking data was generated at various periapsis altitudes. Based on circularization times and final apoapsis altitudes, a periapsis altitude of 51.8 nautical miles was chosen for the aerobraking process; this altitude allows a minimum propulsive burn to raise the apoapsis following aerobraking. In addition, the maximum drag is encountered when passing through this periapsis. The data generated by the analysis for various altitudes are shown in Table 6; the dashed lines indicate that aerobraking is impossible for the respective altitudes. The ΔV s and travel times are presented in Table 7.

Hypersonic Flow Impingement

One important concern of the aerobraking process is the possibility that the hypersonic flow will turn back on the

vehicle, thus causing the flow to strike the vehicle. This impingement problem is a leading factor for determination of the brake diameter; a proper diameter will insure that impingement will not occur. Unfortunately, present research has not developed any procedure to determine the flow reaction during aerobraking. Future analyses using computational fluid dynamics and other computer routines will be required. .

OBSERVATION SATELLITE

One of the important features of the MEPS program is the observation of the Martian surface and atmosphere. Although several spacecraft have flown by the red planet (Mariners 4 and 9) and two have actually landed (Vikings 1 and 2), the data thus far acquired about Mars is insufficient for manned mission planning. To alleviate this lack of data the MEPS mission will include a satellite placed in orbit about Mars. The satellite proposed for Mars observation is based on several systems previously used in Earth orbit; these systems include Landsat-D, Nimbus, and Tiros.

Originally, three satellites for Mars orbit were proposed. However, with the decision to use a separate return vehicle the observation systems of the CIC will eliminate the need for multiple satellites. Thus, only one satellite will be placed into an observation orbit.

The satellite under proposal will be designed for a service life of approximately five years; the long life will allow an extended period of observation of Mars without having to send more satellites. The satellite will be placed in an orbit at an altitude of 380 nautical miles above the Martian surface; this altitude is chosen to get maximum benefit from the observation equipment.

Figure 19 shows the configuration of the satellite under proposal. The experiment packages are located underneath the platform of the three-axis stabilized satellite, with the sensors always pointed towards the surface; thus a more detailed study of Mars can be accomplished. Power will be provided by a solar array panel and back-up batteries will be used when the panel is out of sunlight. Communications between the satellite and the CIC and, ultimately, Earth are accomplished via a parabolic antenna attached to the platform. Propulsion, thermal control, flight software, deployment, and mass will also be discussed in the following description.

Experiment Packages

Several objectives have been set for the observation of Mars. To add to the present knowledge, surface mapping will be performed using a camera system similar to the Thematic Mapper (TM) of the Landsat-D. The TM scans a planet's surface in the visible and near infrared bands with high resolution. The images are then sent to ground stations for image reduction and data reduction. (10:359-360)

For the camera system employed by the Mars satellite, the same scanning bands will be used. All images will be stored on the satellite, and at appropriate times the data will be forwarded to the CIC. All data will be stored in a master file located on the CIC and relayed to Earth-based stations upon command from Earth.

Because little is known about the cloud cover of Mars, an experiment from the Nimbus weather satellite is considered. The High Resolution Infrared Radiometer (HRIR) uses imaging radiometers for studying day and night cloud cover; scanning in several spectral intervals provides much data concerning cloud formation and movement. (10:80-81) A derivative of the HRIR will be positioned on the Mars satellite to investigate the clouds that develop near the polar regions, and to help watch dust cloud and dust storm formations.

Two experiment packages for study of the Martian atmosphere will be incorporated into the Mars satellite. The first is the Filter Wedge Spectrometer (FWS); the FWS determines the "vertical distribution and temperature profile of water vapor and carbon dioxide" in the atmosphere. The second experiment is the Satellite Infrared Spectrometer (SIRS); the SIRS measures the distribution and temperature of the atmospheric gases. (10:80) Both of these packages will allow an extensive investigation and a more complete understanding of the atmosphere about Mars.

The last experiment under consideration is the Space Environment Monitor (SEM) from the Tiros project. This experiment is the same as used on the Command Information Center. A second unit will be used to obtain data about the space environment in the observation orbit of 380 nautical miles.

Solar Array Panel

Power requirement is a very important factor in the design of the satellite. The use of batteries is one possibility, but

there are no batteries that will last the required service life of five years in a space environment. A second prospect is the use of solar cells; satellites presently in Earth orbit have as a common element solar array panels.

Initially, two possibilities for a solar array were considered. The first would place the solar cells directly on the satellite platform; all connections would be internal to the platform, and the satellite would have no appendages except a communications antenna. However, the complexity of the system and the required solar array area removed this proposal from further consideration. The second possibility, a solar array panel, was chosen for the satellite.

A solar array panel has been described for the MEPS vehicle; this design will also be used for power input to the satellite. The satellite panel will be constructed of silicon solar cells, with a coating of silicon monoxide for protection. Two 50-AH batteries will be used for power supply during orbital periods out of direct sunlight.

Because of the large size of the array (110 square feet), the panel will be folded during transit to the proper orbit. Deployment of the array has been described for the MEPS array.

Communications

The communications links to be used by the CIC will also be employed by the satellite. This familiarity will insure the success of the data collection by the experiment packages, and of the data transfer between the satellite and the CIC.

The antenna is located at the end of a 4.0 foot long boom; the boom is composed of a two-segment, 5-inch diameter aluminum tube, similar to that of the Landsat-D. The antenna reflector is of the same assemblage as the MEPS antenna--a graphite epoxy sandwich construction, with graphite-epoxy struts for feed supports. (2:136) The antenna will be pointed directly toward the CIC whenever transmissions between the satellite and the CIC are necessary.

Propulsion

Two different propulsion systems must be considered for the satellite. The first is the Payload Assist Module (PAM) that will transfer the satellite from the 270 nautical mile orbit to the final observation orbit. The second system, which will adjust the attitude of the satellite during the mission, is the Reaction Control System (RCS).

For the required altitude change to 380 nautical miles, the PAM must perform two ΔV s. The first ΔV of 138.4 ft/sec will begin the transfer from 270 nautical miles; the second ΔV of 136.64 ft/sec will be applied to bring the satellite from an elliptic transfer orbit to the circular observation orbit. The transfer will require 65.32 lbm of fuel. The PAM will detach from the satellite once the proper orbit is reached; the orbit will be achieved using a Hohmann transfer.

During the lifetime of the satellite, adjustments to the craft's attitude and orbit will be necessary. Hydrazine will be used as the fuel, and the appropriate thrusters will be fired for the desired adjustments in pitch and roll. For maximum control, on-off pulsing will be used during the burns.

Thermal Control

The only thermal control design under consideration is that of the satellite itself. Use of thermal control coatings, louvers, and radiator units are under examination. Each experiment package will be designed with its own thermal control comparable to the original unit.

Thermal control must be accounted for because of the extreme conditions of the satellite's environment. Heat must be radiated on the sunlit side of the satellite to prevent overheating, and heating must be available to warm the region in the dark. For the latter case, radiator units are the candidate to accomplish the task.

To control the heat on the sunlit side, thermal control coatings such as silver teflon, OSR quartz, or white paint are possible candidates; these coatings have a low solar absorptance to emittance ratio. Movable louvers are also under consideration; the louvers would open or close depending on the amount of heat to radiate. (2:154)

Flight Software

Various software packages capable of performing important tasks must be included in the design of the satellite. Attitude

control computations will determine the spacecraft attitude error relative to the desired orientation. Calculations can be made to determine the position and velocity of the satellite, and these calculations can be used for spacecraft orientation. Changes in the orientation of the antenna and the solar array will occur throughout the mission, and reliable software must be developed to insure that all data and power necessary for the success of the mission is obtained. In addition, the telemetry and the power to the subsystems must be monitored. (2:98-101)

MEPS Deployment

The satellite will be positioned in the same module as the CIC; this positioning will allow system checks during transit to be made easily. The array panel and the antenna will both be folded up.

After the CIC makes preparations to cast off the satellite small explosive charges will be set off. The charges will release springs that will deploy the satellite from the module bay, but the discharge is not sufficiently strong to endanger the delicate CIC systems in the module. At an appropriate distance from MEPS the payload assist module will fire and transfer the satellite to the observation orbit.

Mass Estimates

Data concerning individual components of satellites is not readily available. However, research into entire systems did

yield values to make initial approximations. From this research, the mass of the MEPS satellite, including payload assist module and solar array, is approximately 3500 lbm.

MARS SURFACE EXPLORATION SYSTEM (MSES)

The Mars Surface Exploration System (MSES) has been designed for two purposes: to obtain scientific data about the geology, atmospheric chemistry, and meteorology of Mars, and to carry soil, rock, and atmospheric samples from the surface of Mars to a space station in low Earth orbit. MSES consists of two vehicles that will land on Mars; one vehicle will land on the north polar ice cap of Mars, and the other will land near the Martian equator. Each of these vehicles will carry two surface rovers that will gather soil and atmosphere samples and autonomously explore the surface of Mars in the vicinity of the lander. Also aboard each lander will be an automated laboratory to examine samples retrieved by the rovers, and a sample return vehicle (SRV). The SRV is a solid rocket booster that will carry a payload of Martian surface samples to a low polar Martian orbit (for eventual transit to Earth).

The original design for MSES called for two vehicles to land at each surface exploration site. One vehicle was to carry the SRV to the surface, and the other vehicle was to carry two rovers and an automated laboratory to a point on the surface near the SRV. Due to the difficulty in insuring that the two landers could set down on Mars' surface near each other, the two landers at either site were merged into a single vehicle. This change reduces the total mass of the MSES portion of MEPS because the

SRV, rovers, and laboratory can be protected from atmospheric entry by a single aeroshell that is less massive than the two aeroshells initially required.

The polar and equatorial landers (see Figure 20) are each thirty feet in length. Each has an upper aeroshell that is a cone frustrum twenty-five feet in height, with a twenty-five foot lower base diameter and a seven foot upper base diameter. The lower aeroshell is a spherical segment with a segment length of five feet and a twenty-five foot base diameter. Each lander contains a 14,700 lb, 23 foot tall SRV, two 2,500 lb rovers, deployment ramps for the rovers, an automated laboratory, two robot arms, and a recovery system comprised of solid rocket motors and parachutes.

Transit from Earth to Mars and Insertion into Martian Orbit

In the original proposed configuration of MEPS, the two pairs of landers were to travel with the main spacecraft from Earth orbit, undergo aerobraking into a near-equatorial Martian orbit with the main spacecraft, and then, once the desired orbit had been achieved, depart for landing sites near the equator and at the north pole.

The amount of propellant required to insert the polar lander into a polar or near-polar orbit about Mars from a near-equatorial orbit about Mars was found to be prohibitively large, and so a new trajectory was needed to allow the polar lander to arrive at the north pole of Mars. The mass of propellant required to change the inclination angle of an orbit varies

directly with the magnitude of the change in inclination angle, and also varies directly with the magnitude of the velocity of the vehicle before the burn. The rocket burn to place the polar lander in an orbit that passes over the poles of Mars must be made sufficiently far from Mars to allow the amount of propellant consumed to be minimum. In the limiting case, the position on the trajectory of MEPS (a Hohmann transfer orbit about the sun) that allows a minimum change in inclination angle is at the periapsis of Earth's orbit around the sun, where MEPS' trajectory begins. However, the velocity of MEPS (and therefore of the polar lander) is at its maximum value on the transfer orbit at that location, which is also the periapsis of the Hohmann transfer orbit. The rocket burn to change the inclination of the polar lander's orbit must be made at the position along the Hohmann transfer orbit where the velocity and change in inclination angle allow the rocket burn to be minimum. For a first approximation in calculations, this position will be located at point along the orbit where the true anomaly (angle from orbit periapsis to present position) is ninety degrees.

Thus, at some point along the transfer orbit the polar lander will separate from the main MEPS spacecraft. This lander will be attached to an orbital transfer vehicle (OTV) that will have three stages. The first stage will perform a burn to insert the polar lander (and other two stages of the OTV) into an elliptical polar orbit with a 270 nautical mile perigee over the north pole of Mars. An elliptical orbit was selected over a

circular orbit to conserve mass of propellant that would leave Earth orbit: the estimated mass of the first stage of the OTV is 45,000 pounds. A retro-firing of attitude-control rockets on this stage will cause the stage to enter the Martian atmosphere and burn up.

The equatorial lander will arrive at Mars at nearly the same time as the polar lander arrives and will aerobrake with the main MEPS spacecraft. This maneuver will last on the order of several weeks. After the aerobrake maneuver has ended, the equatorial lander will be released from the end of MEPS.

Atmospheric Entry and Landing

After the polar lander/OTV combination has established the desired polar orbit, the polar lander will detach from the final two stages of the OTV, which will remain in orbit, and enter the Martian atmosphere. The equatorial lander will separate from MEPS after the aerobrake maneuver has been completed; to this lander will be attached a small orbital maneuvering vehicle that will fire to slow the lander down so that it may enter the atmosphere. The density of the atmosphere of Mars is thin, but at high altitudes the lander's velocity will still be very high; aerodynamic heating is therefore expected. For example, at an altitude of 100 nautical miles, the velocity of the lander will be approximately 11,744 ft/sec, and the density of the atmosphere is 3.759×10^{-7} lb/ft³. The temperature on the bottom of the lander is calculated to be roughly 1,630° Rankine. Thus, a

material that is very heat-resistant and dissipative (such as carbon-carbon composite) is required to be on the lower aeroshell of the lander (see Figure 20).

The lander decelerates during the descent through the atmosphere. Deployment of a pilot parachute (11.7 feet in diameter), then a drogue parachute (54.4 feet in diameter), and finally three main parachutes (129.7 feet in diameter) will slow the lander to a terminal velocity of 75 feet per second. At this point, the lower aeroshell is separated from the to allow the four landing gear to deploy.

At an altitude of 210 feet from the surface of Mars, the upper aeroshell (to which the parachutes are attached) will detach from the lander and an array of small rockets beneath the upper aeroshell (inside the lander) will carry the shell away from the remainder of the lander. At this instant, an array of solid rocket motors with a net thrust of 100,000 pounds will fire, slowing the lander even further. These rockets will allow the lander to descend the final five feet to the surface at a constant speed of ten feet per second. To allow the lander to withstand the impact of touchdown and to compensate for uneven terrain, the landing gear will be hydraulically adjustable by a computer on board the lander. Also, since surface dust will be disturbed during the landing, and might settle on the sensitive machines that are on the lander, a post-landing inspection (via

camera on the robot arm) will be carried out. A cleaning mechanism such as compressed gas should be designed for the robot arm to "dust off" the ship.

After the upper aeroshell has been removed from the lander, the rovers, laboratory, and SRV will be exposed. Upon touchdown, the surface exploration phase of the mission begins. Two ramps will extend from the lander platform to the ground, allowing the two surface rovers to descend and begin collection of data and samples.

Rover Design Parameters

As previously described, two rovers will be located in each lander. The lander which touches down near the pole will contain the Polar ice Cap Rovers (PCR). The second lander, which will land near the equator, will contain the Equatorial Rover (EQR). Each rover will weigh approximately 2500 pounds. This weight was originally based on the Viking weight of 1320 pounds on the Martian surface; the weight was increased due to the addition of tracks for mobility, equipment for gathering samples, and artificial hardware and software for complete automation. With these added accessories the rovers will be eight feet long and four feet wide (see Figure 21 and 22). An artist's conception of the rover collecting samples is shown in Figure 23.

Rover Design Changes

Originally, the EQR was designed to be mounted on four separate tracks for mobility; this design would enable the rovers to have more maneuverability. The PCR was proposed with a sled/ski

configuration on the front and two tracks on the back; the sled/ski would provide smoother travel across the icy surface. After further study the two additional tracks on the EQR and the sled/ski on the PCR were found to be more complicated than necessary to carry out the mission. High maneuverability can be achieved with just two tracks equipped with hydraulic motors for separate steering. The PCR will have the same two tracks as the EQR but will be equipped with spikes to grip the icy surface.

Power

Several systems have been considered for the power systems of the rovers. Radioisotope thermoelectric generators (RTGs) produce energy from the heat released by the decay of radioactive fuel; these generators were used on the Viking, but reasonable doubt exists concerning the use of RTGs to supply the power required by the rovers. Side-mounted solar panels could provide adequate auxiliary power, but dust storms will constrain the amount of available energy. Once the power requirements for each rover system are defined, detailed analyses can be used for proper selection of the power source.

Traverse Cycle

Each rover will be capable of traveling three tenths of a mile each Martian day, which is forty minutes longer than an Earth day. A total distance of two hundred miles will be covered each Martian Year, or 1.88 Earth years. The amount of ground covered by the rovers seems relatively small, but the process as a whole must be considered. The traverse cycle, the

travel from one destination to another, alone will be approximately a forty minute process. The first ten minutes will consist of viewing the travel area with a laser scanner for any foreseeable danger. The next five minutes will involve the planning of a feasible route to the next site of exploration; this planning will be accomplished by a microcomputer. The actual transit will encompass the remaining twenty-five minutes. With a twenty-five minute period for travel the rover will cover approximately two hundred and fifty feet each traverse cycle, and six to eight traverse cycles will be completed each day. The remainder of the day will be spent gathering the required samples at each site.

Landing Site Location

The location of the landing sites is based on the availability of unique geological and atmospheric conditions. The equatorial landing site is planned for nine degrees South latitude and one hundred and forty-four degrees longitude; this site was previously approved for the Viking mission. (6:303) The location of this site is ideal for geological sampling and Olympus Mons is within viewing range for further observations.

The location of a polar landing site is a more complicated decision. The north pole was selected due to its diversity in meteorological activities. The exact longitude has not been decided upon due to the lack of information and pictures of the northern region. The extension of the northern ice cap during the winter is known to be much less than the south polar region,

normally advancing slightly below sixty degrees north latitude. (26:111-8) With the proposed transfer orbit MEPS will be arriving at Mars in late September of 1999, at which time the Northern hemisphere will be experiencing spring. The conditions will be ideal for landing above sixty degrees north latitude, and the rover will travel northward following the dwindling glaciers of the polar ice cap. Traversing this zone of just-melted ice may yield signs of plant life. The spring season will last for 199 days, allowing the rover enough time to retreat southward before the ice cap begins extending in the fall. Once the ice cap has fully extended the rover will begin the process of mapping the extension.

Mission Definition

The mission definition for each rover will vary depending on the site selection; each rover will have the same basic mission with different variations for the different regions. Three basic missions will consist of searching for life and obtaining core samples, seismometer readings, geochemical data and climate data.

The search for life involves acquiring soil samples which will be analyzed by the lander laboratory. Viking did not find any signs of life for the duration of its mission; although highly unlikely that any form of life will be found by the rovers, more elaborate tests might result in different findings. The soil samples will be taken using a shovel adapter for one of the robotic arms. This arm will have several different adapters for digging, drilling and crushing. The other robotic arm will

have the capability to lift and grip objects. Once collected, the samples will be placed in canisters located on the top of the rover. These canisters will be removed from the rovers by the robot arms of the lander upon return to the lander.

Core sampling will also be used to determine if life was or is in existence on Mars. A rotary-driven auger adapter will be used to extract these samples, which will have to be large enough to encompass anticipated textural inhomogeneities. Different sizes of augers will be required for different types of formations; therefore, the samples will also range in size. The largest will not exceed one and a half inches in diameter and one and a half feet in length. The number of core samples gathered will be limited because of their size. The SRV will only be capable of transporting approximately eight core samples, which limits the rovers to four core samples a piece. These samples will need to be stored in a cold area in the event the samples contain volatile materials.

In order to obtain these core samples, the drilling arm will require a supply of compressed carbon dioxide. The carbon dioxide will be used to cool the drill bits and to clean out the hole for extraction of samples. The PCR drilling arm will required a reversed carbon dioxide fluid circulation to clean out the holes being drilled through the permafrost. (3:1)

"Seismometer data reflects motion in three directions. In addition to telling scientists the magnitude and duration of seismic activity, the instrument will provide information on the

source of activity." (26:111-69) The two seismometers installed on the Viking were never successful. One was damaged upon landing, and the second seismometer could not make accurate readings because of its location on top of the lander. For successful readings the seismometers will be erected on the surface and left to function autonomously.

Each rover will assemble one seismometer using the two robotic arms. Data will be triangulated to determine precisely where a seismic event occurs and variations can be compared to determine the nature of the planet's structure. The signals produced will be converted to digital data which will be relayed to the CIC.

The collection of atmospheric data will be performed by a meteorology boom similar to the boom used on the Viking. The meteorology instruments will periodically measure pressure, temperature, and winds during each Martian day. This data will be compared to the testing site area, and orbital photography to better understand Martian weather and atmosphere. Area photographs will be taken by a camera system on board the rovers. This system will consist of three separate cameras; two of the cameras are designed from the Viking mission, and the other camera will act as a guidance system for transit. The orbital photography will be performed by the CIC and the satellite.

Communications

The CIC will act as a data link with Earth for the rovers. At preset times when the CIC passes overhead, collected data will

be transmitted from the rover. This data will eventually be relayed to Earth ground stations to be reduced by Earth scientists.

The CIC will also relay messages from Earth. If the satellite or CIC has transmitted a photograph with interesting characteristics that need to be observed, the rover will be relayed the location and desired samples to collect. After the SRV has returned to Earth and more scientific tests have been run on the samples, more samples may need to be taken. For this reason the CIC will have continued communications with the rovers after the samples are returned.

Lander Laboratory

Tests will be conducted in the lander laboratory before the samples are returned to Earth. The canisters of samples located on the top of each rover will be unloaded by the robotic arms. From this point on the lander will be in charge of carrying out the prescribed experiments.

The lander lab will contain various instruments for biological investigations. A pyrolytic release experiment will search for signs of micro-organisms that function by photosynthesis and chemotrophy. A labeled-release experiment will search for signs of metabolic activity in the soil samples. The gas-exchange experiment will indicate any gaseous changes which symbolize life processes are going on. (26:111-62,63)

The biological experiments can only determine if active life processes exist. Thus, the organic investigation experiment will

be useful in determining if life forms ever existed in the past, or have the capability of existence in the future. The Gas Chromatograph Mass Spectrometer (GCMS) will conduct the organic investigations. The GCMS passes the solid samples through several furnaces and a chromatographer for vaporization and separation of the different organic molecules. The organic compound vapors are then ionized in the mass spectrometer to produce a profile of each compound. (26:111-64)

To make sure that investigations of the samples on the Space Station will not infect any scientist, the samples will be examined in the lander lab. The X-ray Fluorescence Spectrometer (XFS) will give a chemical breakdown of the samples. Each sample will be charged with photons, which "causes a transfer of energy to the atoms in the sample's basic elements and causes the atoms to expel electrons". (26:111-66) The vacancy caused by the electron is filled by another electron, thus creating a release of energy. The amount of energy released characterizes the basic chemical element which enables the XFS to identify the element.

Transportation of Mars Samples to Earth

Periodically, the rovers will return to the lander to have samples which are contained in sealed cylindrical metal containers unloaded by the two robot arms that are stationed upon the platform near the SRV. These robot arms will be twenty-five feet long when fully extended. After examination in the automated lander laboratory, the samples will be resealed in their containers and placed inside the payload capsule (Figure 24) of

the sample return vehicle. The payload capsule will have a hatch that can be easily opened and closed by the robot arms. The hatch will also be capable of being sealed with the turning of an outer latch by the manipulator of a robot arm. The interior of the payload capsule will include a refrigerated cabinet for storage of sample containers, along with guidance and control computers for the ascent and insertion into the 270 nautical mile circular orbit of the SRV.

Preparations will begin for launch of the SRV when the sample cabinet in the payload capsule of the SRV is filled. The rovers will move several hundred yards from the lander platform, to avoid being damaged by the exhaust of the SRV. The laboratory will be heavily shielded on all sides, and all openings to the laboratory will be sealed by the robot arms prior to launch. The robot arms will be retracted into a protective hatch during the launch of the SRV, since they are located on the platform beside the SRV.

The command for the SRV to launch will be transmitted to a receiver in the payload capsule of the SRV. This launch command signal will have been relayed to the launch site from Mission Control on Earth via the CIC. Mission Control will be able to monitor the launch, but with a time delay of roughly fifteen minutes real-time adjustments by Mission Control will be impossible during the SRV ascent trajectory. Thus, the launch sequence for the SRV will be completely automated. The guidance system of the OTV will be contacted by Mission Control, also via

the CIC. This contact will activate the OTV, which will have remained in its elliptic polar orbit during the stay of the polar and equatorial landers on the surface of Mars.

Each SRV will consist of a single-stage solid rocket motor and its payload, and will be propelled into a 270 nautical mile circular polar orbit. The payload consists of the payload capsule (which contains the Mars samples), a reaction control system, and a conical aeroshell. The launch of each SRV must be timed such that the rotation of Mars brings the longitude of the lander site directly beneath the polar orbit of the remaining two stages of the OTV. The SRV burn time is approximately 336 seconds, and the final acceleration of the SRV is 3 g's.

When a 270 nautical mile altitude has been reached, the payload stage detaches from the booster stage, rotates such that the aeroshell faces away from the direction of motion, and detaches from the aeroshell. This rotation exposes the reaction control system of the payload (see Figure 24). The trajectory of each payload capsule during launch from the surface is monitored by the CIC. The orbit of the payload capsule is adjusted based on "advisories" from the CIC to the payload capsule, which fires its reaction control rockets as directed by the CIC or by Mission Control.

The OTV second stage, an orbital maneuvering system controlled by the guidance computer in the third stage of the OTV, makes a series of small burns to rendezvous with the payload capsule. The OTV will maneuver to intercept the payload capsule,

which will detach its reaction control system, leaving only the central capsule that will contain the samples. The OTV will have to be highly automated to make the tricky rendezvous required to capture the 500 pound central capsule. This operation must be performed for both SRVs so the final payload that the OTV will transport from Mars to Earth is 1000 pounds.

After both SRVs have been captured, the second stage of the OTV will detach from the OTV, leaving the third and final OTV stage and its payload of 1000 pounds. The third OTV stage, which has a mass of approximately 4500 pounds, performs a burn to take the samples from the Martian polar orbit into a transfer orbit about the sun to return to Earth. A second burn will be employed at or near Earth to decelerate the vehicle to approach the velocity of the Earth relative to the sun. At Earth a vehicle must rendezvous with and attach to the returning spacecraft and perform a burn to place the OTV third stage and the two payload capsules containing the Martian samples into Earth orbit.

ESTIMATES

Two estimations must be made to complete the MEPS design study. The time estimate and cost estimate will now be discussed.

Time Estimate

The time required for the final design preparation, construction, and mission of MEPS has been estimated and is presented in Table 9. The estimate for the mission is based on the proposal to reach Mars in 1999. Using a profile for the 1999 conjunction, requiring a launch date in late 1998, the projected time necessary for the completion of MEPS construction and the Mars mission is presented. (25:108)

Cost Estimate

The estimated cost of MEPS (see Table 10) has been determined from an estimate for a previously proposed manned mission. This manned mission is scheduled for the same time period as the MEPS mission. Only costs pertaining to unmanned systems have been considered. (12:941)

SUMMARY

An unmanned probe mission to Mars is considered as the appropriate beginning to the proposed manned Mars missions of the early 21st century. As defined in this final report, the Multiple Exploratory Probe System can lay the foundation for future manned missions by increasing the present knowledge of the planet Mars; the scientific areas that will benefit include atmospheric and surface studies and aerobraking analysis. Although the major emphasis of the mission is placed on the data from surface rovers, an orbiting satellite, and various other relays, the success of the mission depends on the central computing unit (Command Information Center), the propulsion stages, the guidance and control, and the Earth-Mars transfer.

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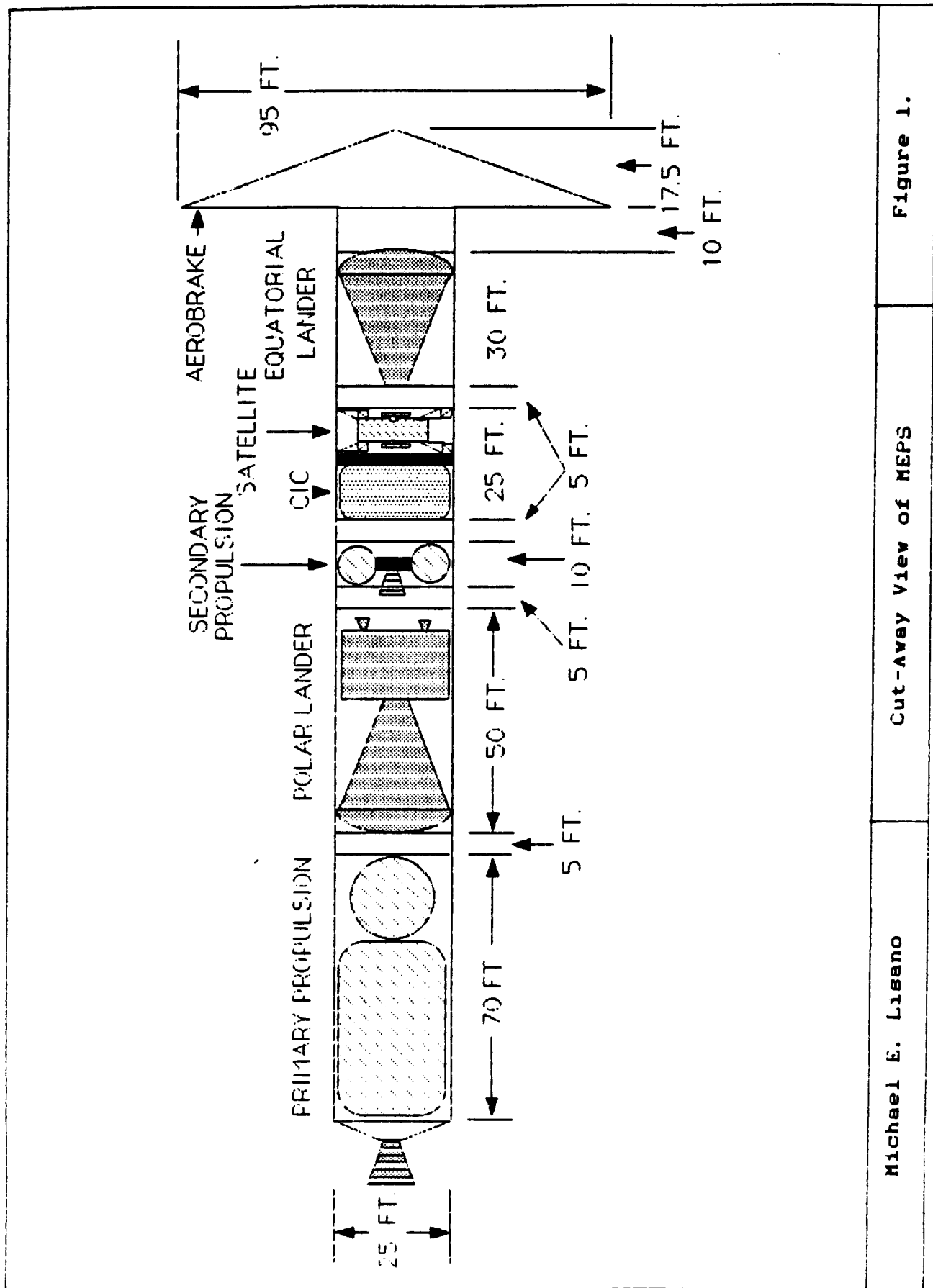


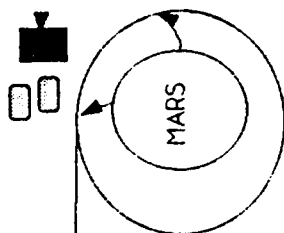
Figure 1.

Cut-Away View of MEPS

Michael E. Lisano

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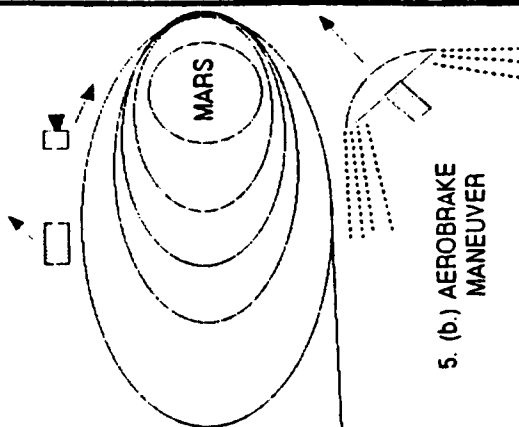
6. OTV CAPTURES
SAMPLE CABINETS



7. OTV/SAMPLES RETURN TO EARTH

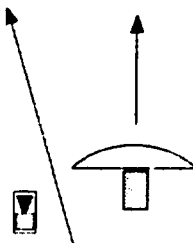


5. (a.) DESCENT OF POLAR LANDER

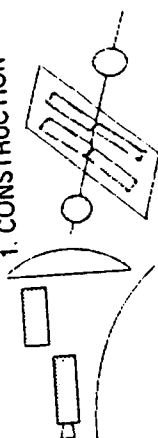


5. (b.) AEROBRAKE
MANEUVER

4. DEPARTURE OF POLAR LANDER



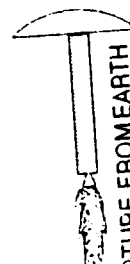
1. CONSTRUCTION



3. SEPARATION OF
PRIMARY ENGINE



2. DEPARTURE FROM EARTH

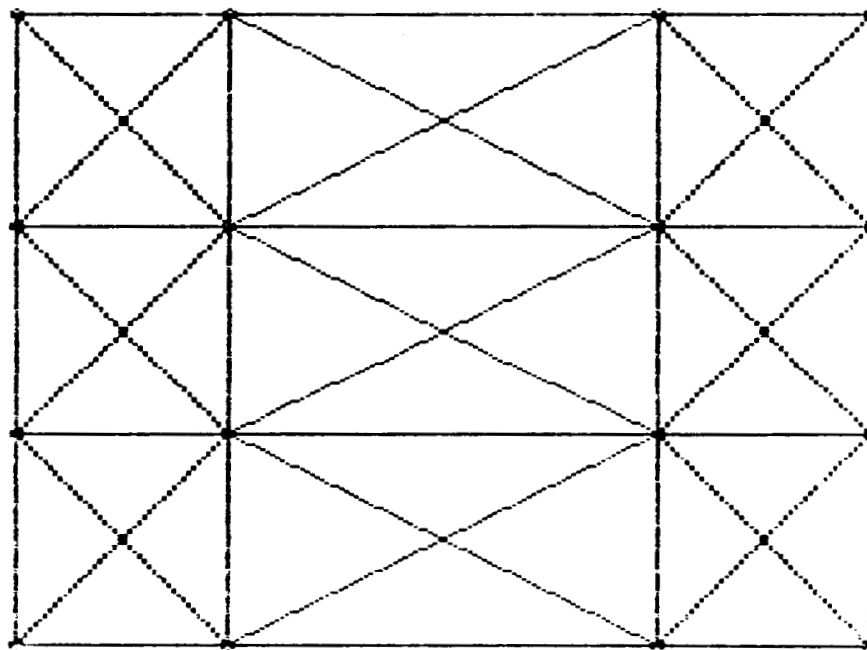


Profile of the MEPS Mission--
Earth-Mars and Sample Return Transit

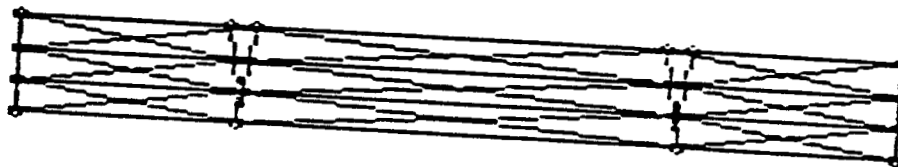
Michael E. Lisano

Figure 2.

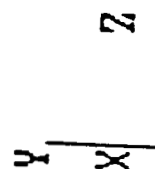
Undeformed



Deformed



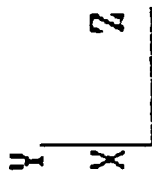
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Scott A. Striepe

Back Slash Truss Deflection--
Deformed and Undeformed Shapes

Figure 3.



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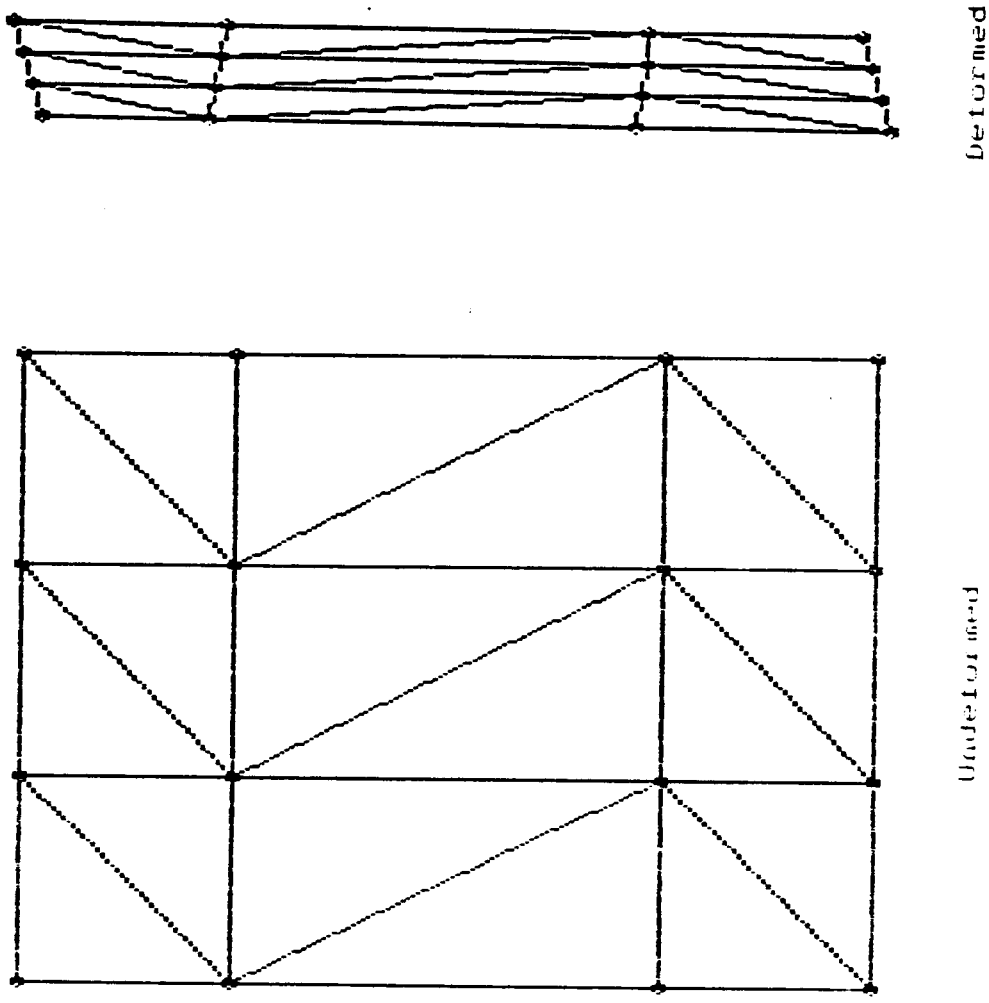
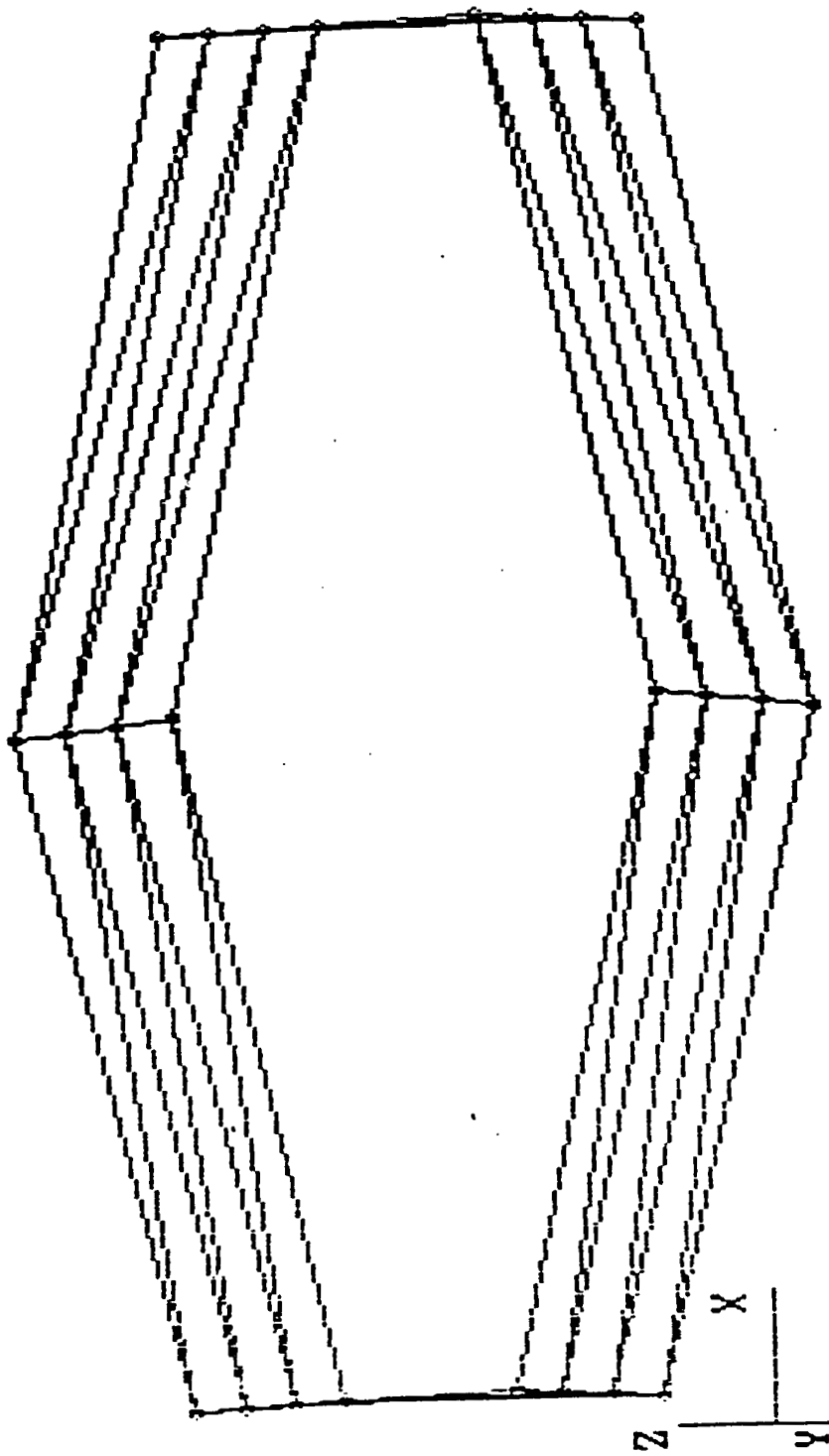


Figure 4.

Alternating Slash Truss Deflection--
Deformed and Undeformed Shapes

Scott A. Striepe



Scott A. Striepe

Back Slash Configuration
After Rotation

Figure 5.

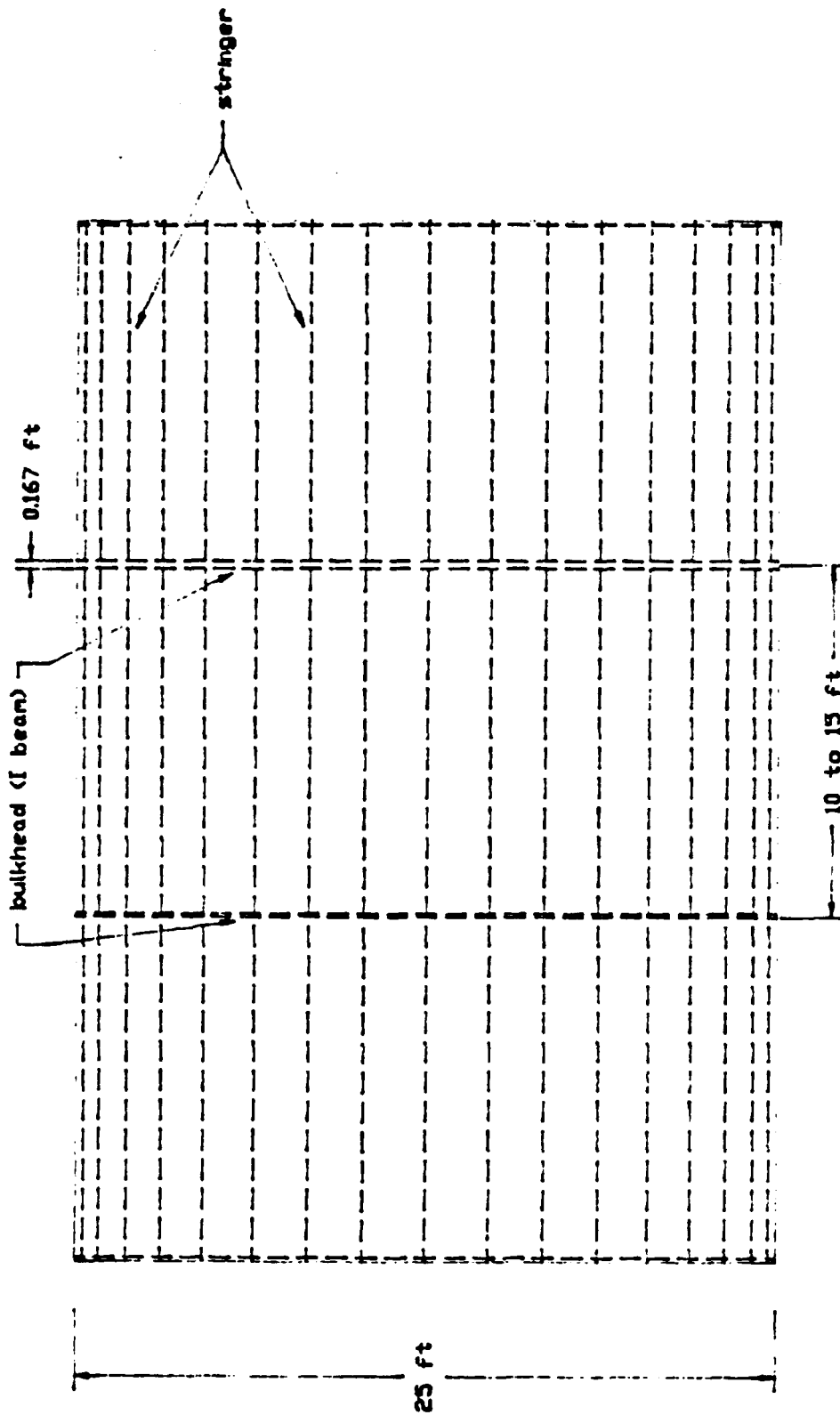


Figure 6.

Example Module (Side View)

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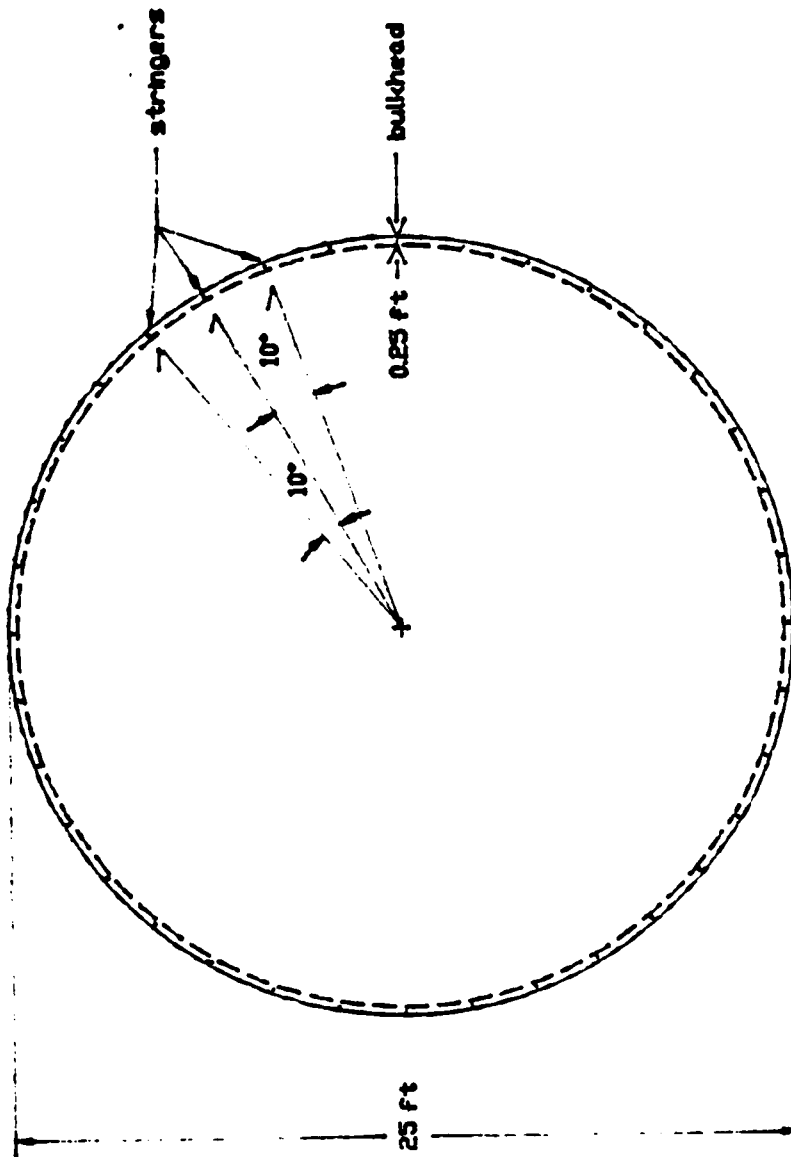


Figure 7.

Example Module (Rear View)

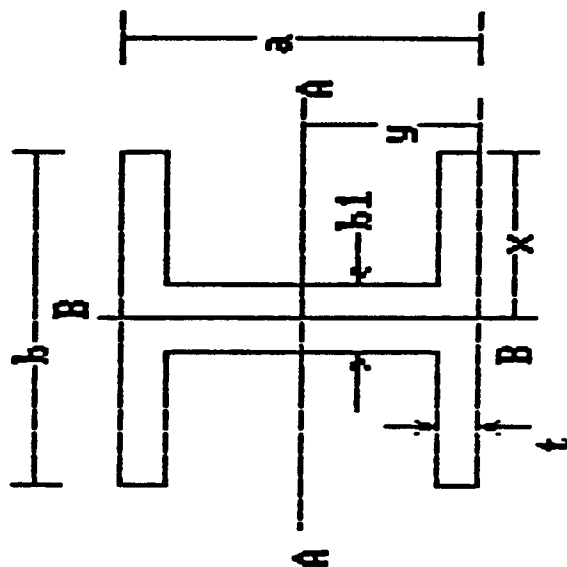
Scott A. Striepe

Module A-1: Properties of Plane Sections

DIMENSION a = 3.
 DIMENSION b = 2.
 DIMENSION b1 = .1
 THICKNESS t = .25

Area	=	1.2500E+00
y	=	1.5000E+00
x	=	1.0000E+00
I _{aa}	=	2.0260E+00
I _{bb}	=	3.3354E-01
r _a	=	1.2731E+00
r _b	=	5.1656E-01
C _w	=	6.3021E-01
J	=	2.8383E-02
K _a	=	1.9992E-01
K _b	=	6.6640E-01

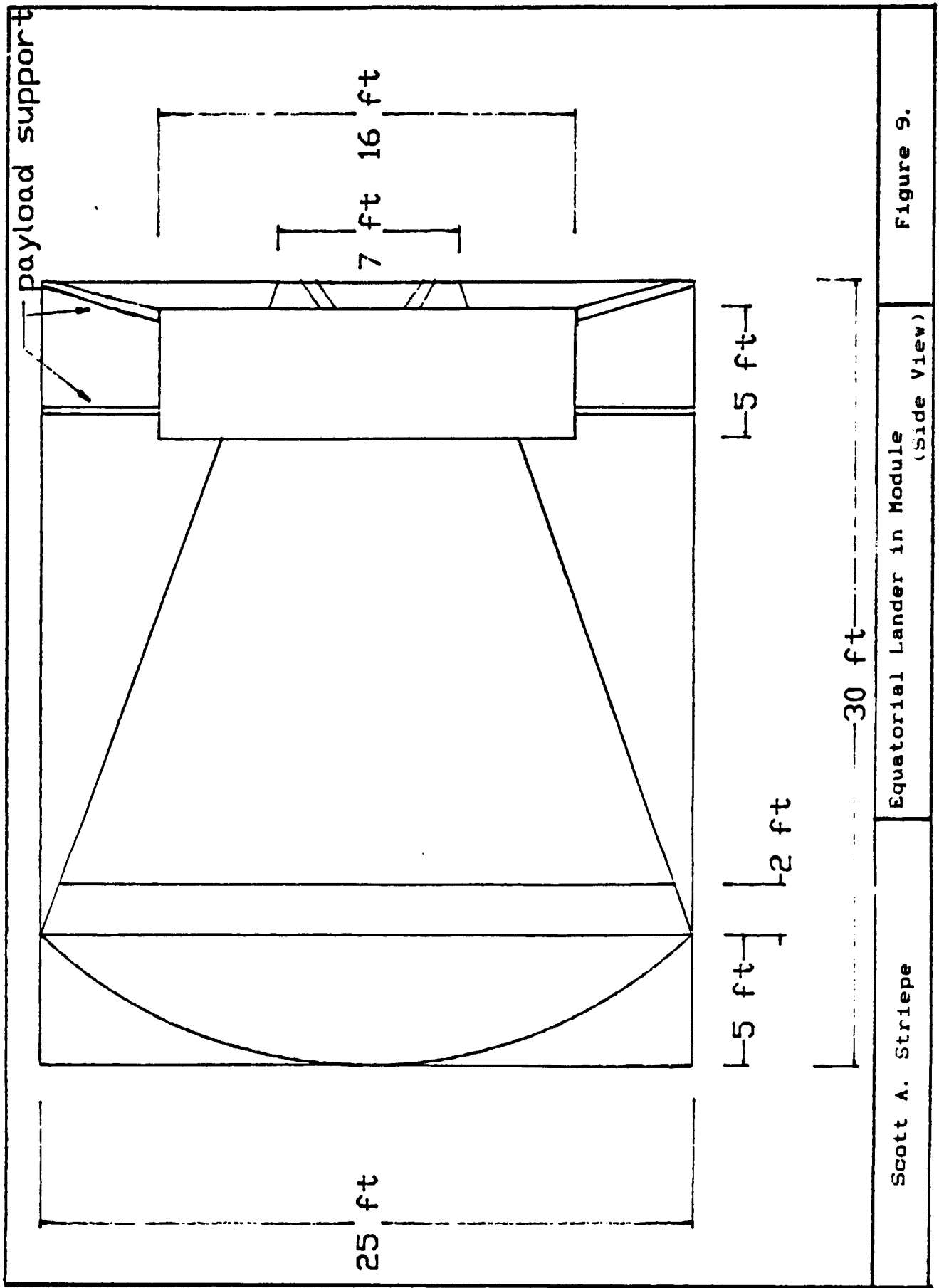
23. Symmetrical I-Section

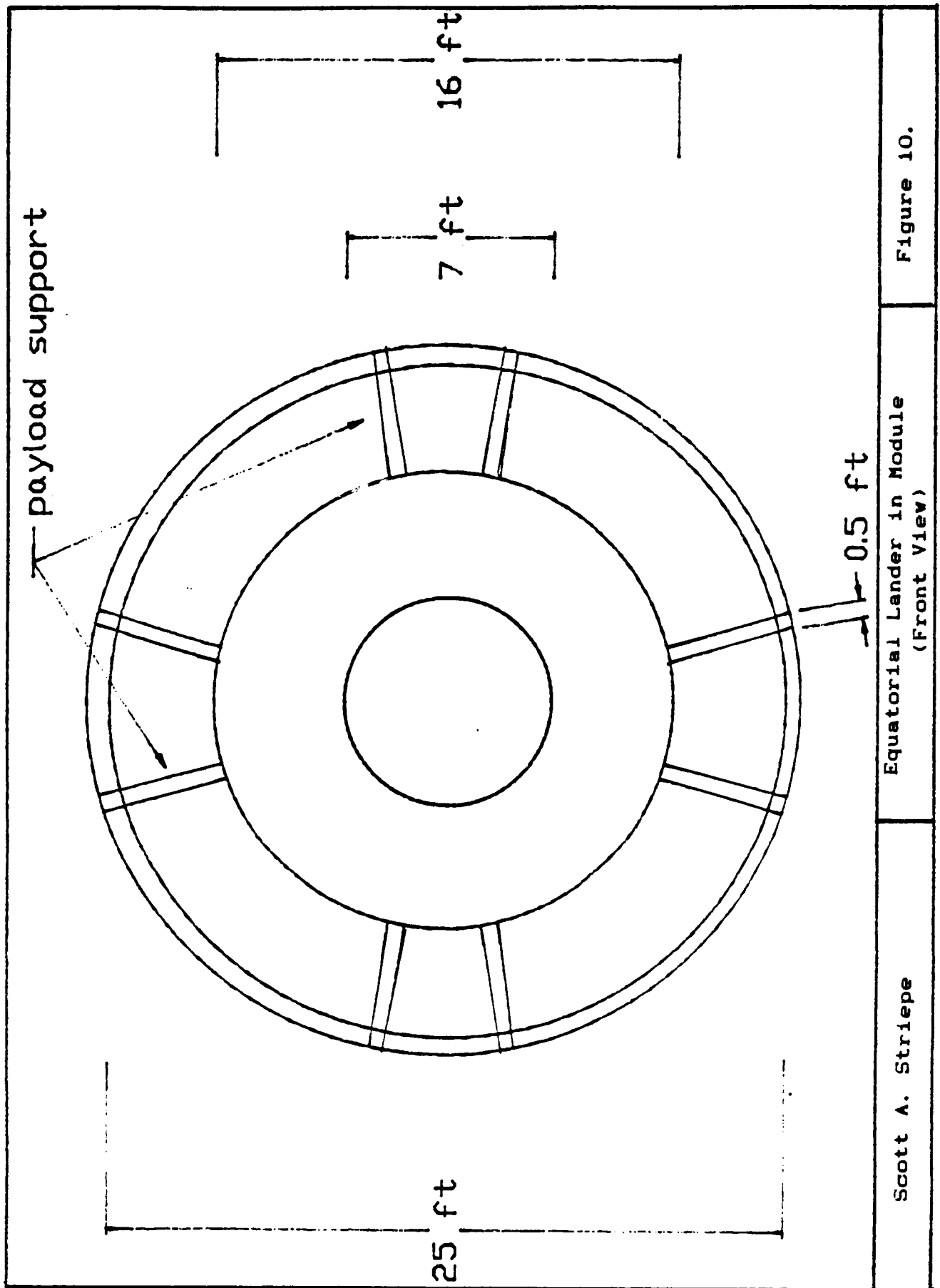


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Bulkhead Cross Section Properties

Figure 8.





MSC/pal 2

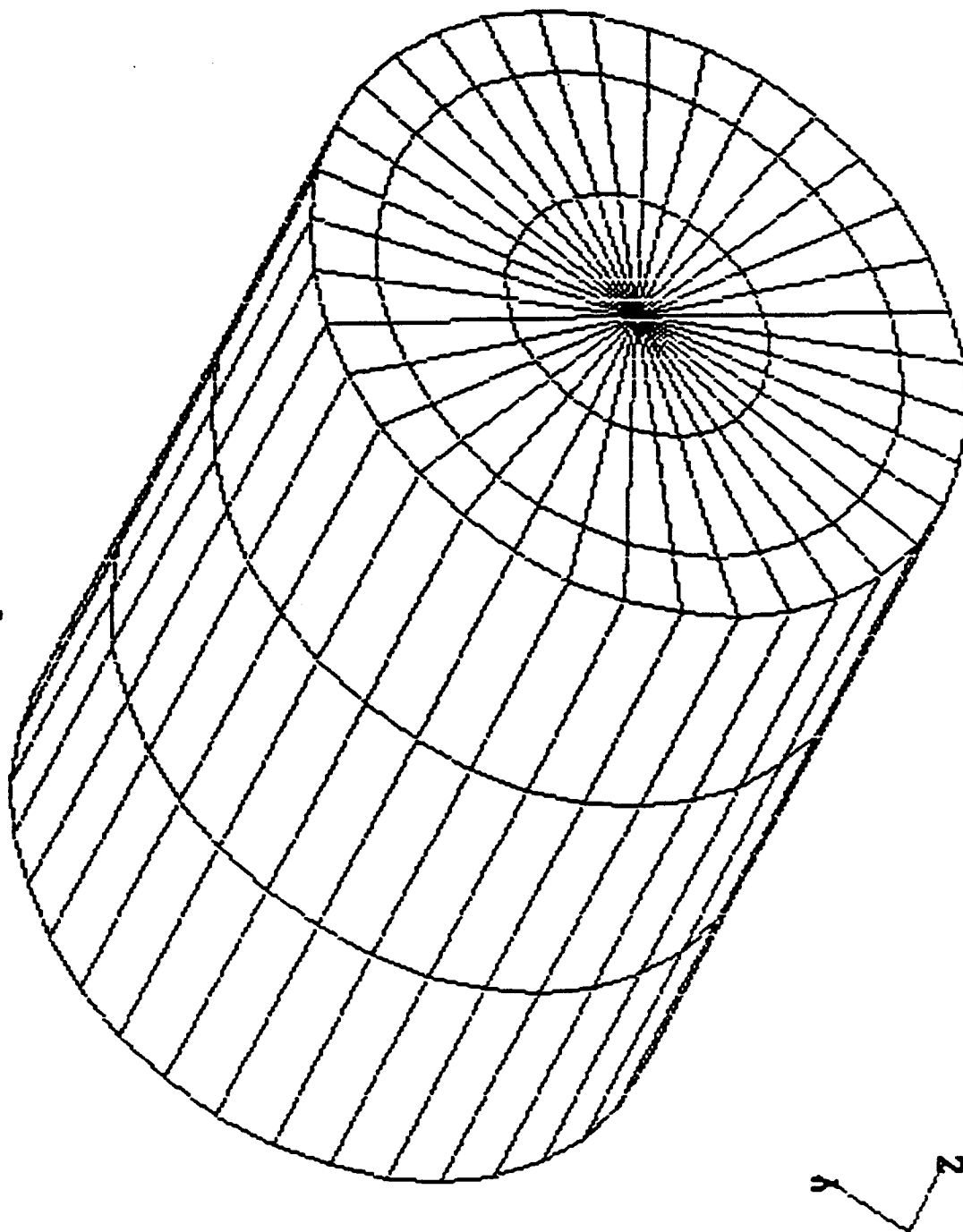


Figure 11.

Finite Element Cylinder

Scott A. Striepe

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MSC/pal 2

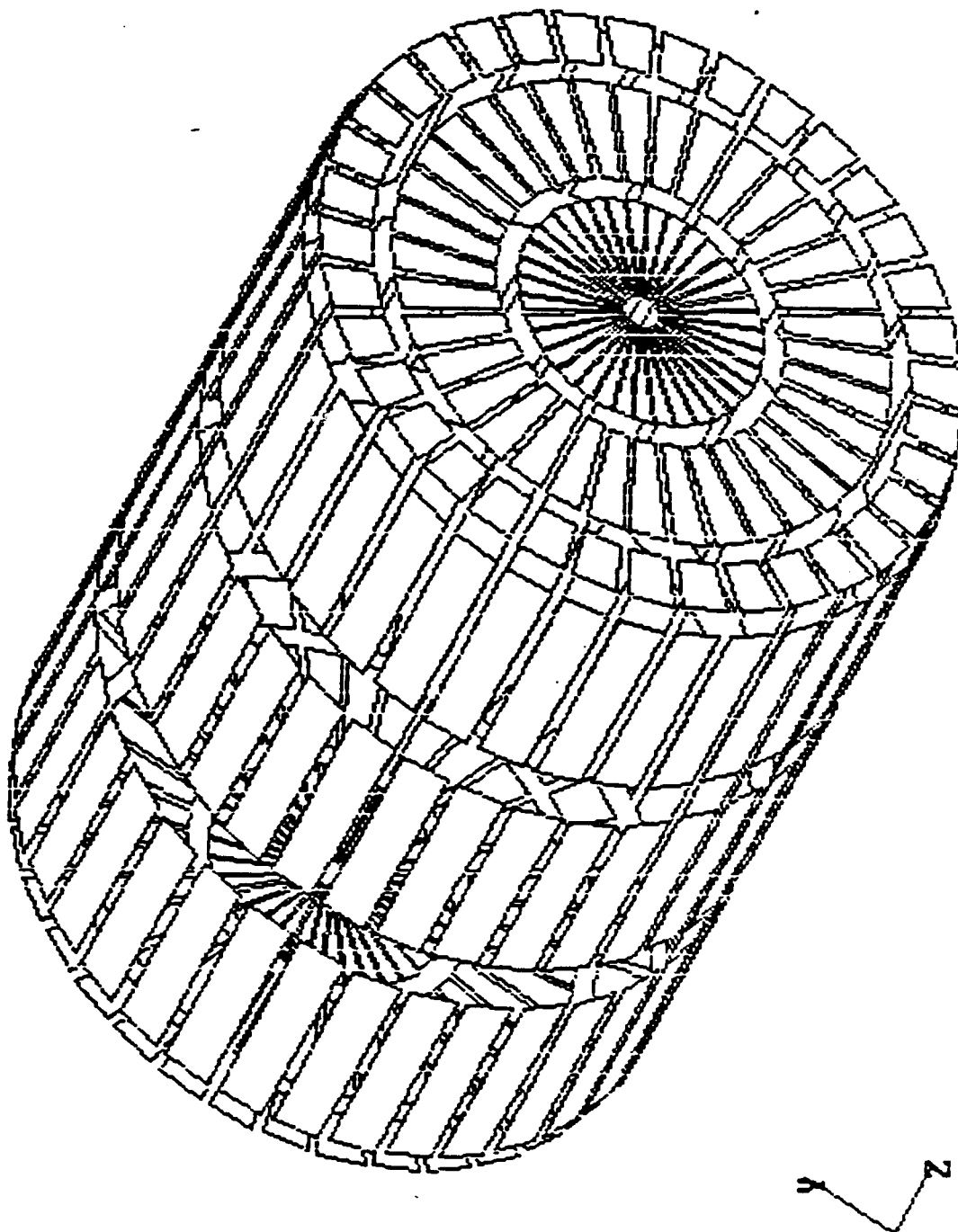


Figure 12.

Finite Element Cylinder 20%
Shrinkage

Scott A. Striepe

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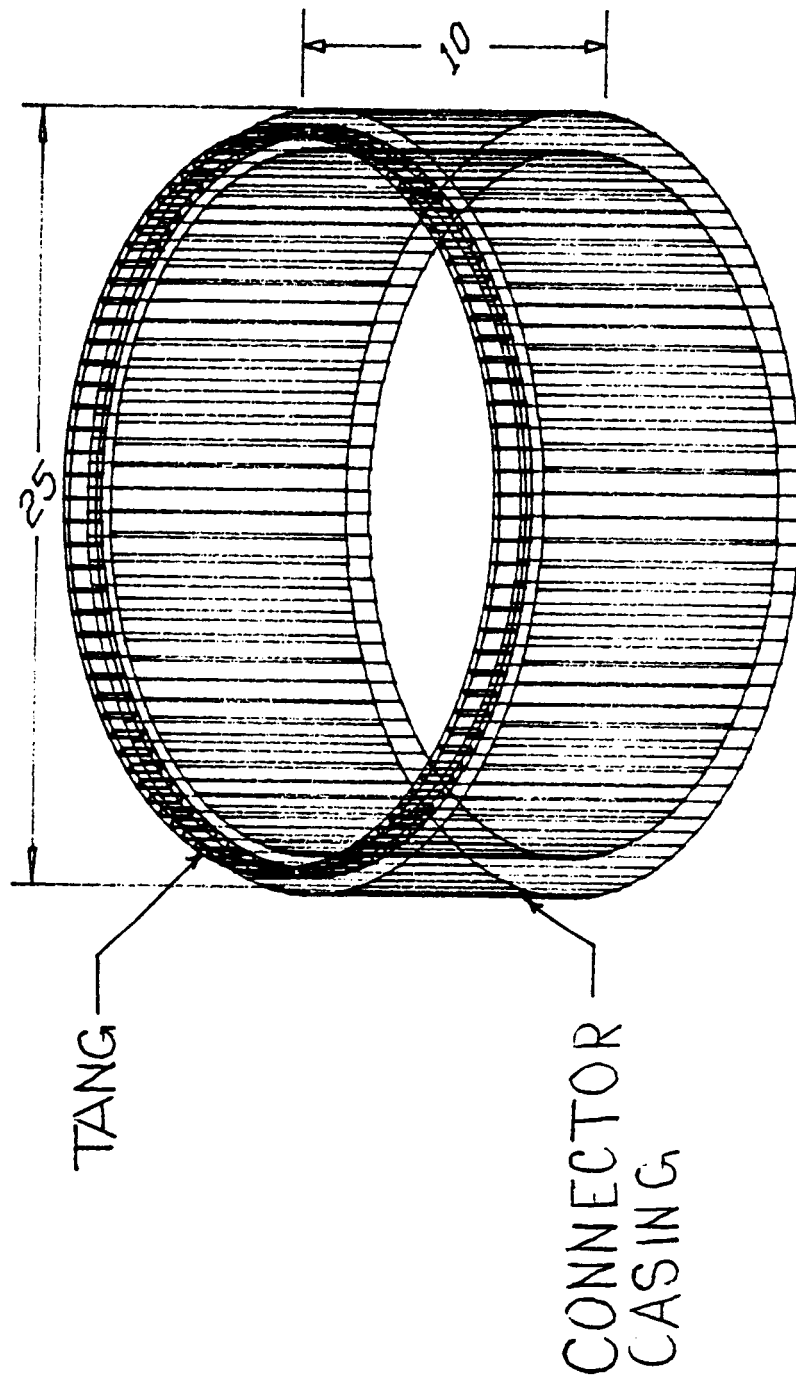


Figure 13.

Connector of Modules

Chris M. Crumbly

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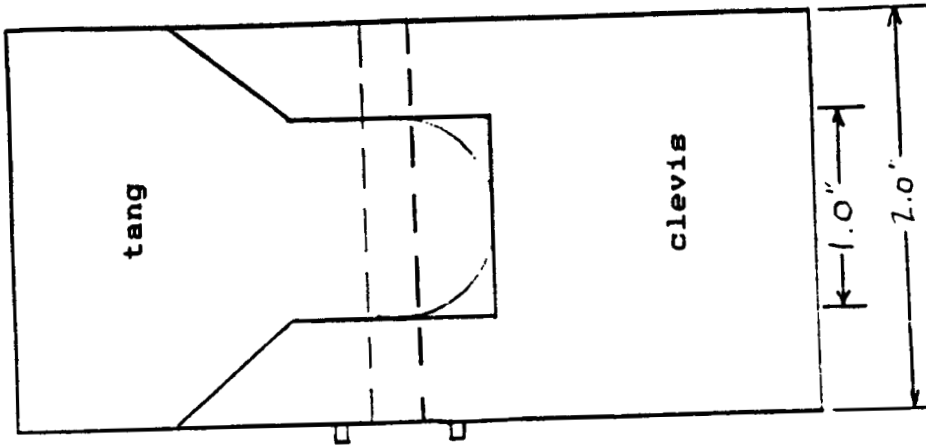
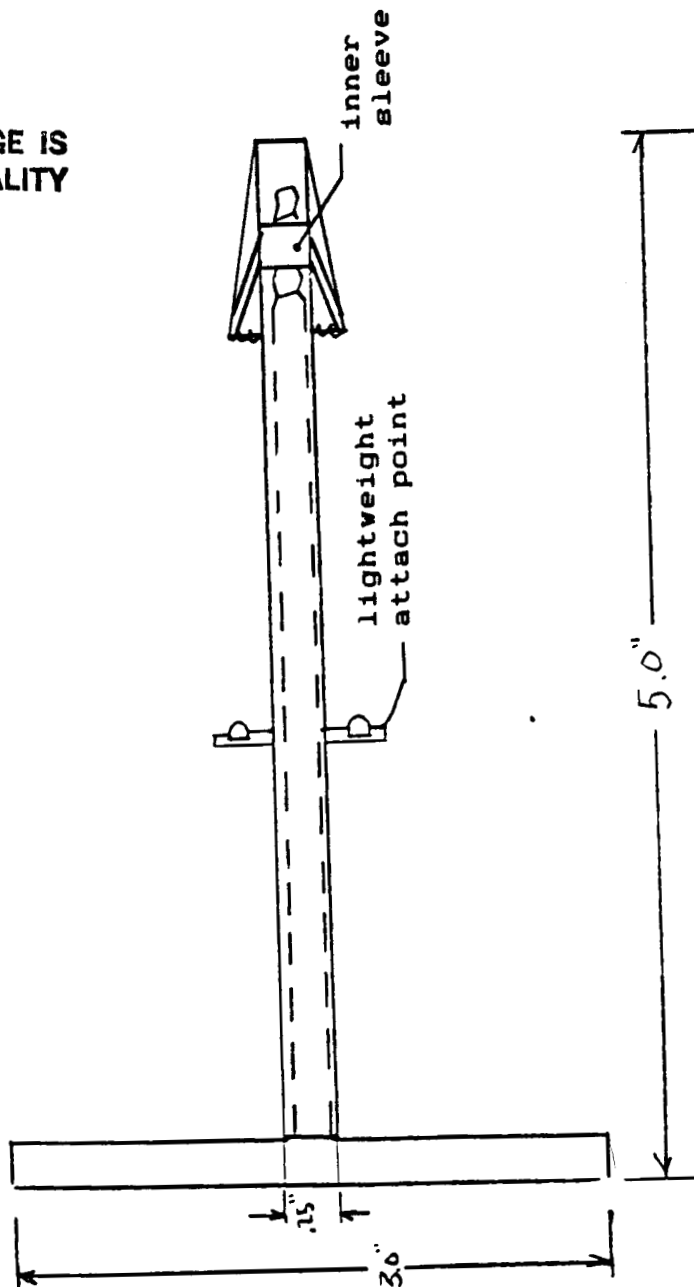
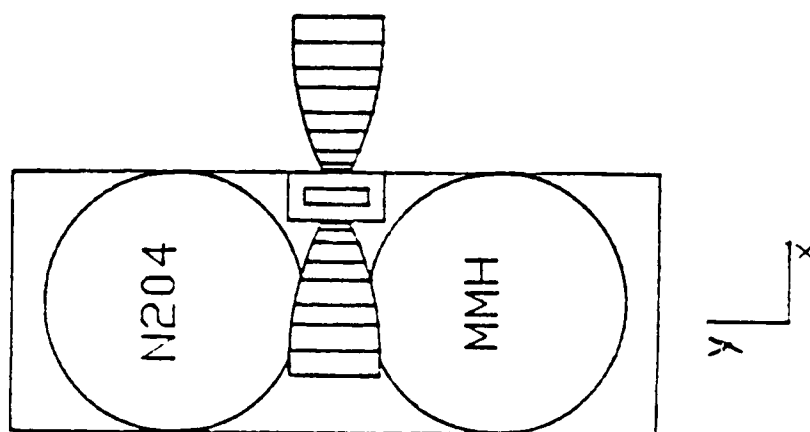
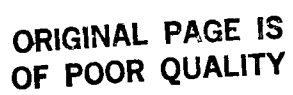


Figure 14.

Attach Pin for Connector

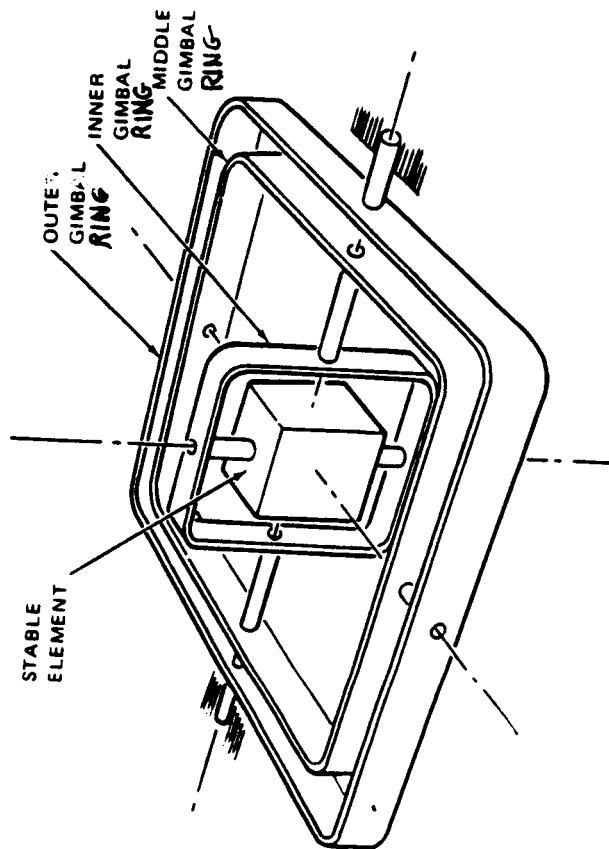
Chris M. Crumbly



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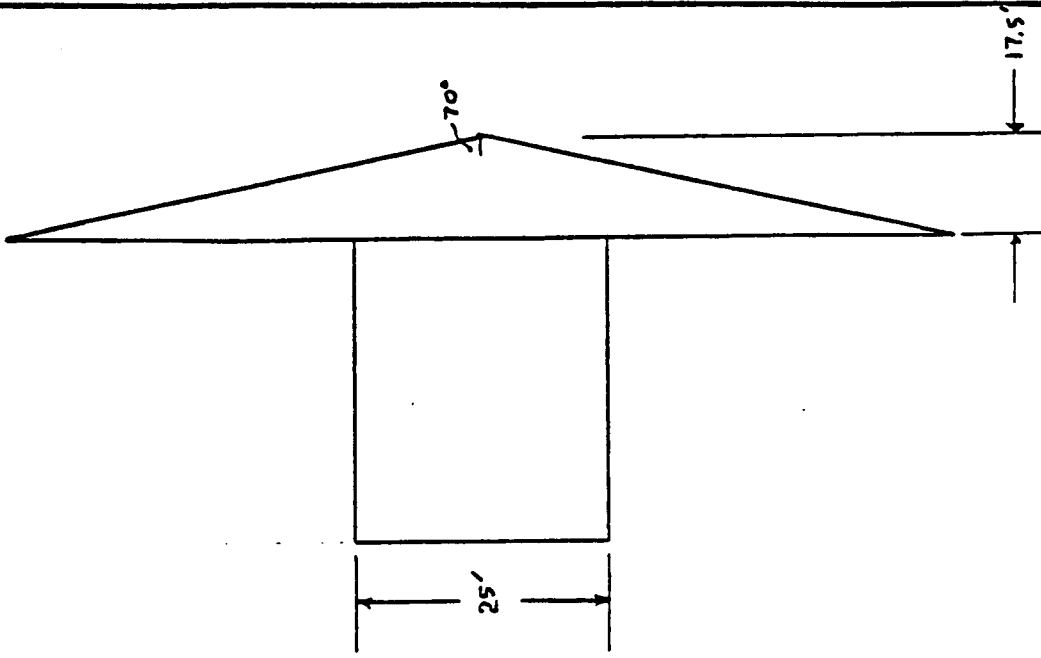
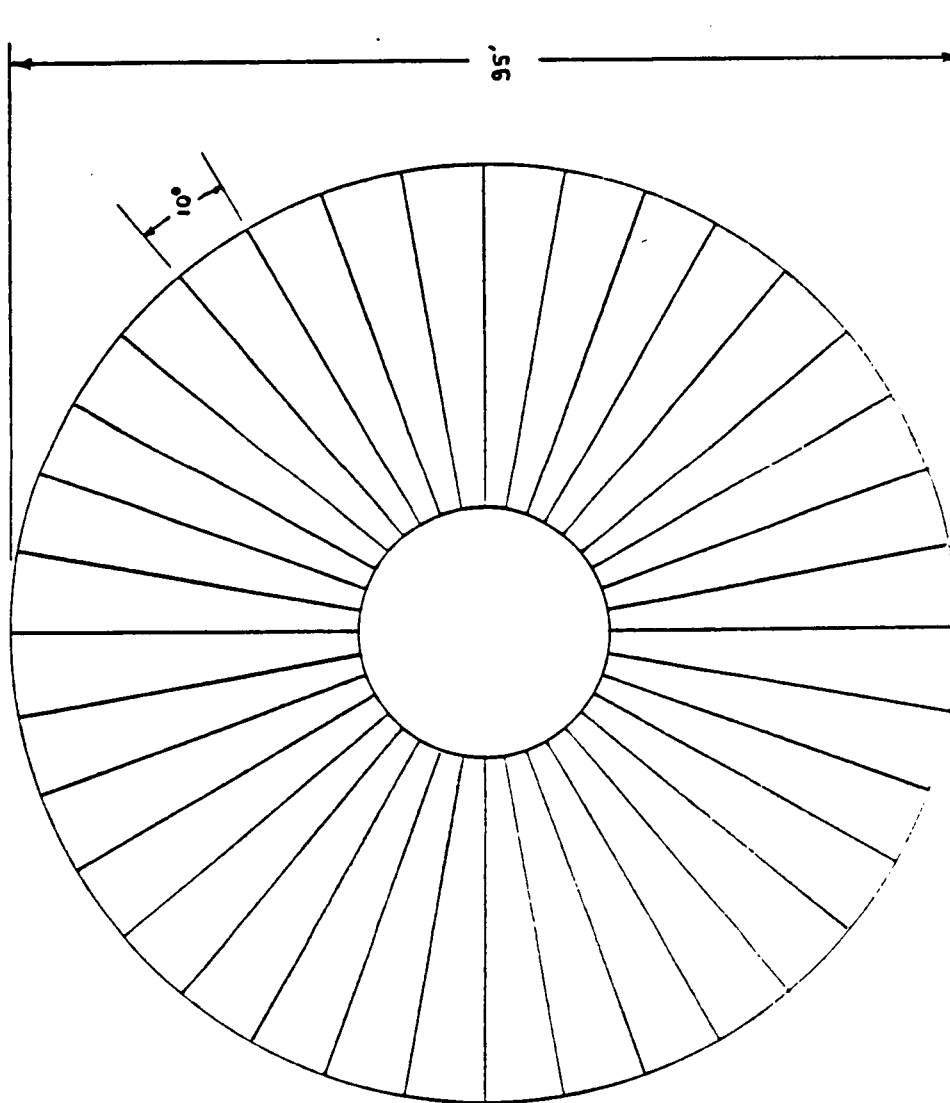
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Platform Gimbal Rings

Air Force Systems Command

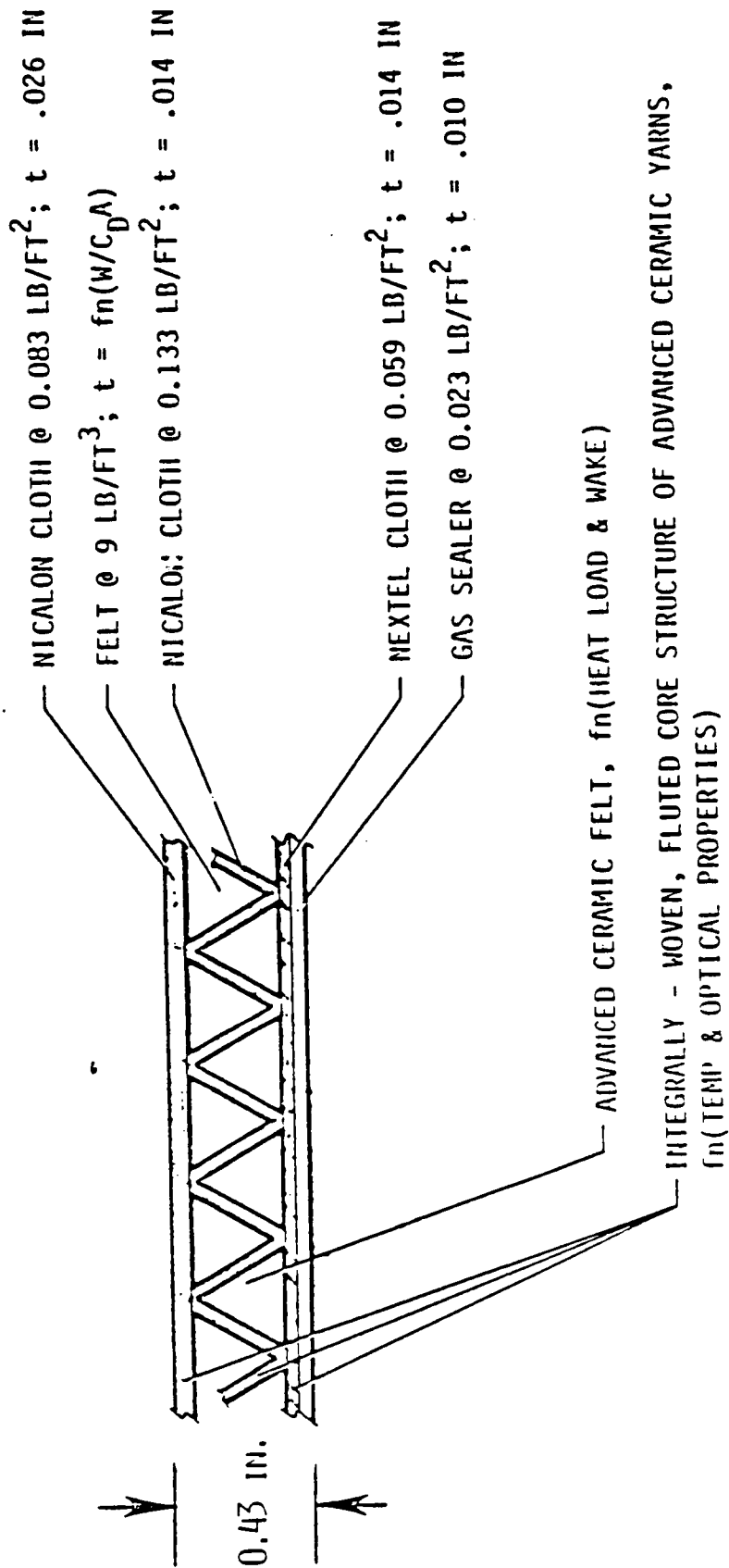
Figure 16.



James D. Packard

View of the Aerobreak for the
MEPS Mission

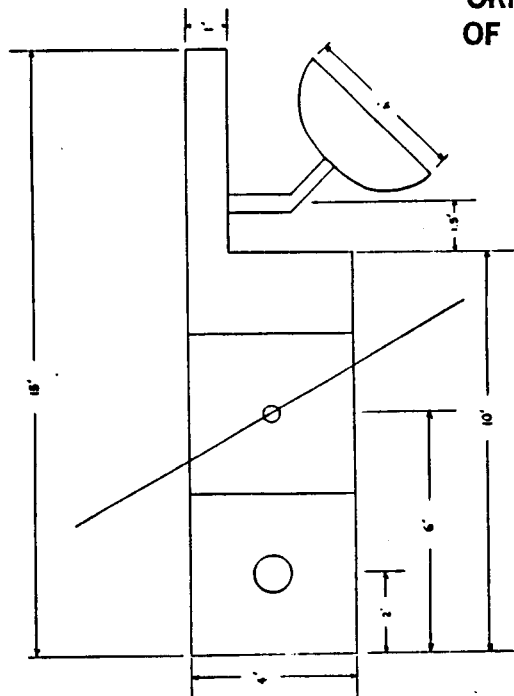
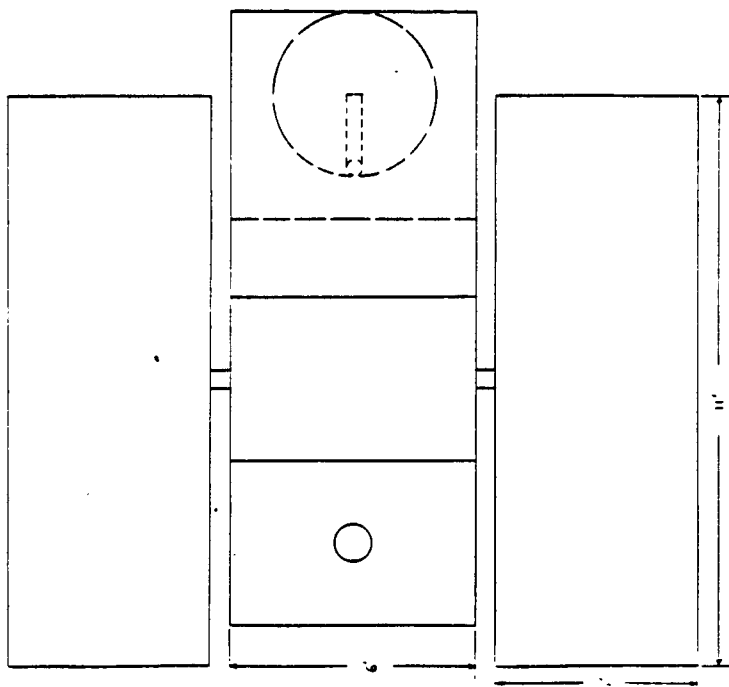
Figure 17.



Martin Marietta

Flexible Thermal Protection System

Figure 18.



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Top View
Panel at 0°, Antenna Undeployed

Side View
Panel at 60°, Antenna Deployed

James D. Packard

Satellite for Mars Observation

Figure 19.

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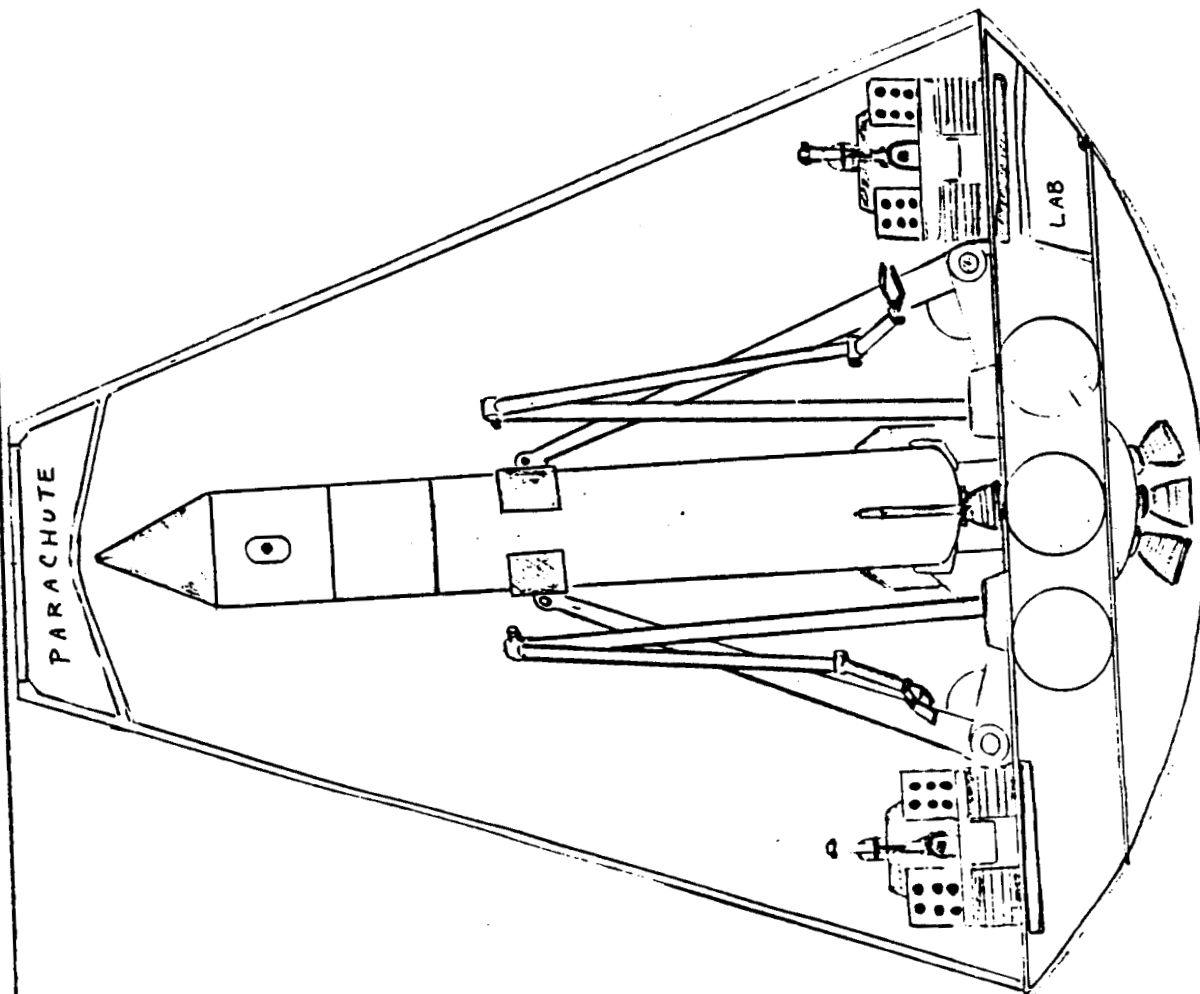
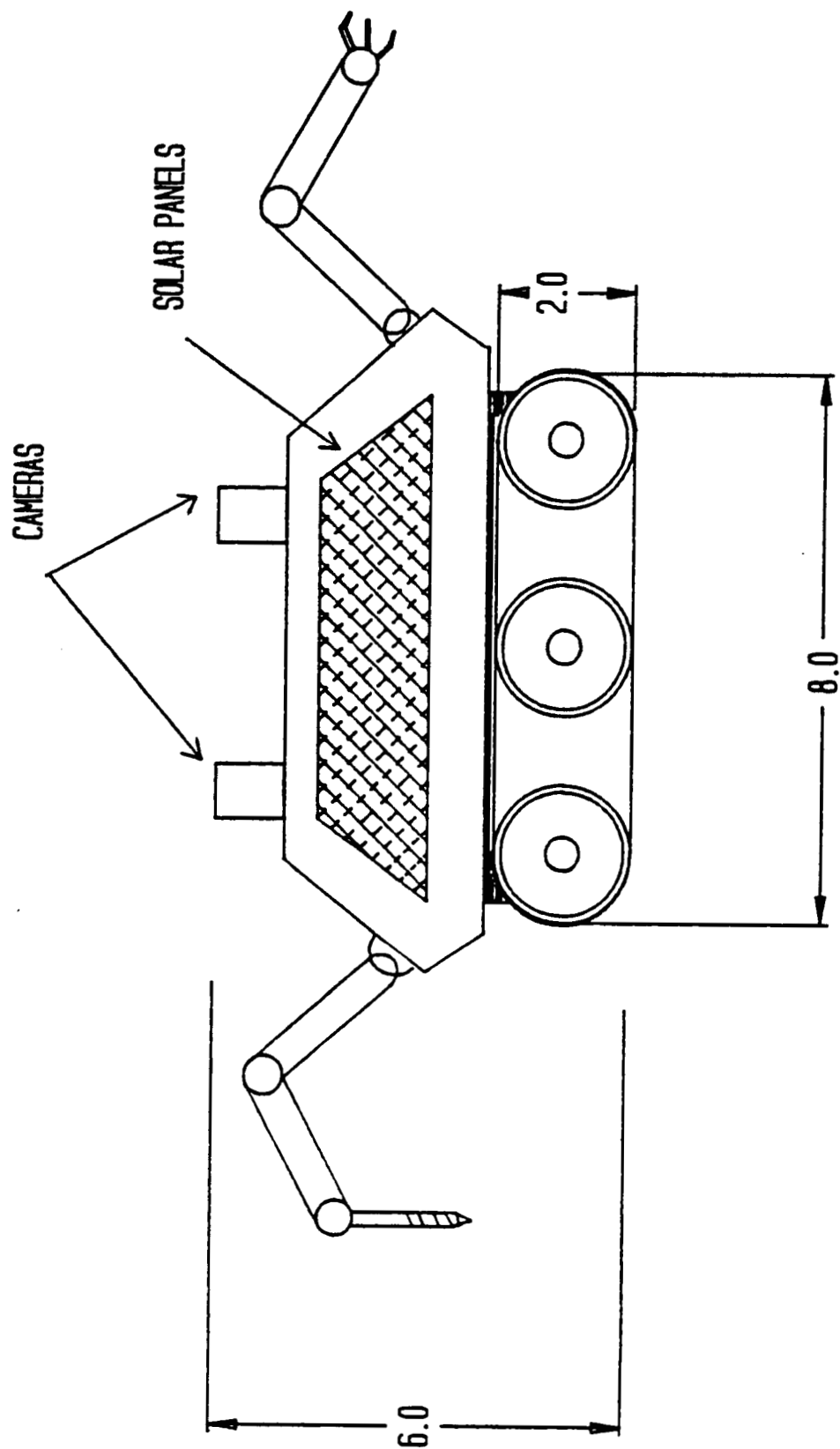


Figure 20.

Cut-away View of a Mars Lander

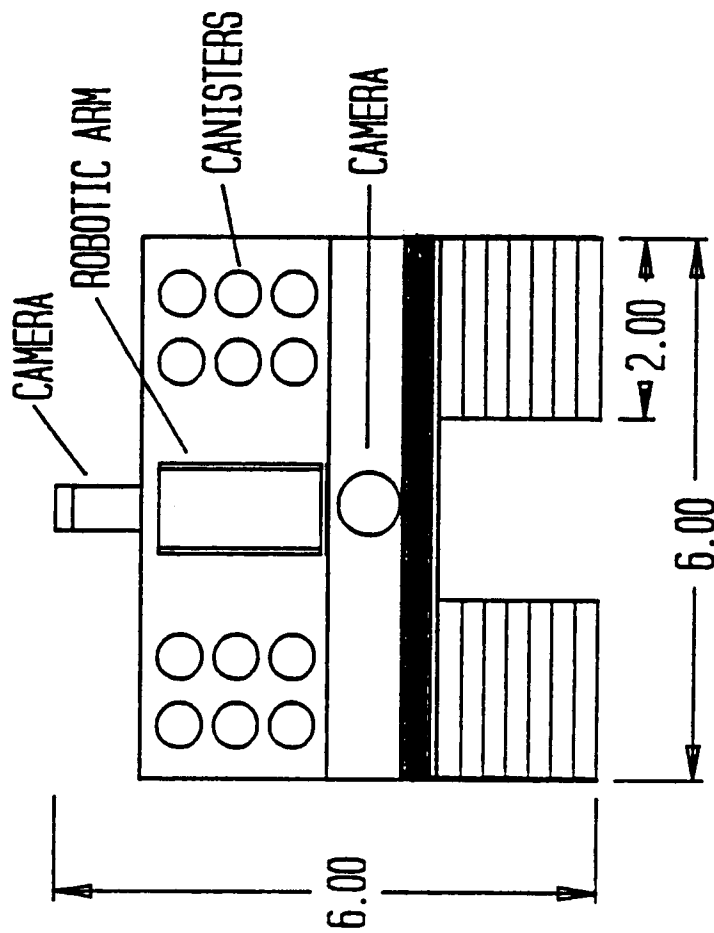
Michael E. Lisano



Michelle A. Guidry

Rover (Side View)

Figure 21.

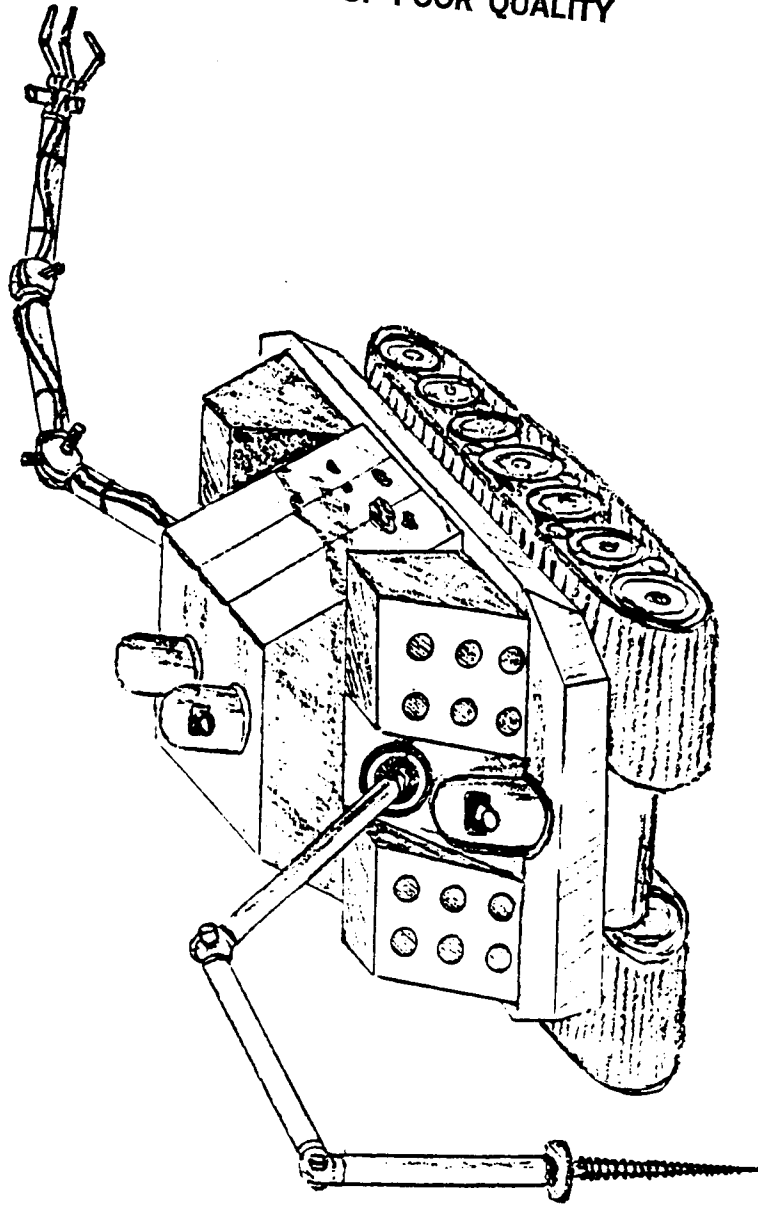


Michelle A. Guidry

Rover (Front View)

Figure 22.

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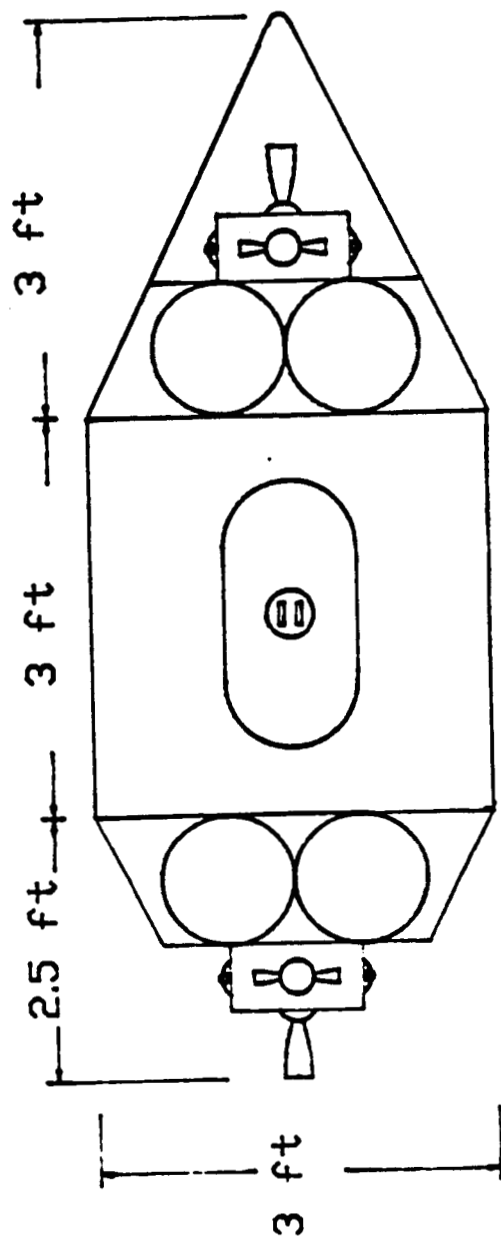


Michael E. Lisano

Artist Conception of a Mars
Surface Rover

Figure 23.

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Michael E. Lisano

Sample Return Vehicle
Payload Capsule

Figure 24.

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Table 1. MEPS Main Vehicle and Length Breakdown

Module	Length (feet)	Structural Mass (lbm)	Non- Structural Mass (lbm)	Total Mass (lbm)
Equatorial Lander	30	13845	65000	78845
Polar Lander with OTV	50	20625	110000	130625
Satellite/ CIC	25	12100	6500	18600
Secondary Propulsion	10	7065	65810	72875
Main Propulsion	70	27055	480000	507055
Non modules				
Aerobrake	18	n/a	12000	12000
Connectors	5 or 10	34050	n/a	34050
TOTAL	215	114740	739310	854050

Table 2. Data for Earth-Mars Transfer

Earth Orbit Altitude	270 n. mi.
Gravitational Parameter, Earth	$1.408(+16) \text{ ft}^3/\text{sec}^2$
Semi-Major Axis of Transfer Ellipse	$1.018(+6) \text{ n. mi.}$
Initial Propulsive Burn	11641.52 ft/sec
Gravitational Parameter, Sun	$4.686(+21) \text{ ft}^3/\text{sec}^2$
Gravitational Parameter, Mars	$1.506(+15) \text{ ft}^3/\text{sec}^2$
Final Propulsive Burn	3048.34 ft/sec
Elliptic Orbit About Mars	(270 x 17788) n. mi.
Time for Earth-Mars Transfer	285 days

Table 3. Parameters of STME

Burntime	401.24 sec
Volume LO_2	4688.3 ft^3 (35133.4 gal)
Volume LH_2	12564.1 ft^3 (94198 gal)
Mass LO_2	333200 lbs
Mass LH_2	55.533 lbs
ISP	449 sec
Thrust	435000 lbs
Weight	7455 lbs
O/F ratio	6.0
Area Ratio	55/141

Table 4. Delta-V Burns for Orbit Circularizations

<u>Burn No.</u>	<u>Delta-V (ft/sec)</u>	<u>Propellant Mass (lbm)</u>
1	3074.90	58850.44
2	75.12	1213.86
3	301.66	4773.69
4	12.11	188.36

Table 5. Design Parameters of the Proposed Aerobrake

Cone Half-Angle	70 degrees
Drag Coefficient	1.766
Ballistic Coefficient	19.3
Weight to be Aerobraked	200,000 lbm
Diameter	95 ft
Number of Ribs	38
RSI Weight	401 lbm
FSI Weight	3850 lbm
Structure Weight	5000 lbm
Aerobrake Total Weight	12,000 lbm

**Table 6. Results of Aerobraking Analysis
at Various Periapsis Altitudes**

Periapsis (n.mi.)	Max. Drag (lb)	Final Apoapsis (n.mi.)	Final Semi- Major Axis (n.mi.)	Time (days)
53.964	217.517	2065.975	1976.175	38.305
51.805	388.908	2092.886	1988.550	21.873
49.647	691.562	2007.292	1944.674	12.947
47.488	1225.779	2032.072	1955.984	7.964
45.330	2182.928	-----	-----	-----
43.171	3877.277	-----	-----	-----

**Table 7. Propulsive Burns and Travel Times
for Aerobraking Process**

Burn to Lower Periapsis (Pre-Aerobraking)	-75.1214 ft/sec
Burn to Raise Periapsis (Post-Aerobraking)	301.6568 ft/sec
Burn to Raise Apoapsis (Post-Aerobraking)	12.1129 ft/sec
Time in Orbit Prior to Aerobraking	48.792 hr
Time for Aerobraking Passage	472.141 hr
Travel Time from Periapsis to Apoapsis (Post-Aerobraking)	0.944 hr
Travel Time from Apoapsis to Periapsis (Following Periapsis Raise)	1.023 hr
Period of Orbit following Circularization	2.053 hr

Table 8. Time Estimate for
MEPS Mission

1987 1988 1989 1990 1991 1992 1993 1994 1995 1996 1997 1998 1999 2000 2001 2002

Design Phase	/-----/														
Development Phase	/-----/														
Testing and Evaluation	/-----/														
Module Fabrication	/-----/														
Operational Space Station															
Construction in LEO	/-----/														
Pre-Launch Tests	/-----/														
Earth to Mars Transit	/-----/														
Mars Observation	/-----/														
Mars to Earth Transit	/-----/														
Data Evaluation	/-----/														

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Table 1. Cost Estimate Analysis
for MEPS Mission

(in billions of 1985 dollars)

Manned mission estimate -----	24.0
Manned systems -----	9.0
 TOTAL COST ESTIMATE -----	 15.0
 STRUCTURES PROGRAM -----	 2.0
truss, nodes	
HLLV deployment	
Shuttle deployment	
outer skin	
 PROPULSION PROGRAM -----	 3.5
unit development	
construction	
propellants	
 ORBITAL INSERTION PROGRAM -----	 4.5
aerobraking	
retro-rockets	
 COMMAND INFORMATION CENTER -----	 2.0
trajectory/guidance and control	
software	
satellite configuration	
development and construction	
Primary Return Vehicle	
 SATELLITE PROGRAM -----	 1.0
development	
construction	
 SURFACE EXPLORATION PROGRAM -----	 2.0
Rover Carrier	
Equatorial Rover	
Polar Cap Rover	
Sample Return Vehicle	
development	